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Title

GENERAL INSTRUMENT INTERFACE CONTROL DOCUMENT

GICD

	Name and Function	Date	Signature
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FOREWORD

The METOP documentation, related to the instrument interfaces consists of the following two documents :

- the General Instrument Interface Control Document (GICD - DRD - 2 1).

  This document aims at defining all the requirements on the interfaces, tests and programme to which all the instruments shall comply for the METOP mission. It is a generic specification, applicable to any of the METOP payload complement instruments, that deals with interfaces from the platform towards the instruments.

- The Instrument Interface Control Document (ICD) Outlines (DRD - 22).

  This document gathers each individual instrument ICD outline that defines the technical and programmatic interface information applicable to a particular instrument. It then deals with interfaces from the instruments towards the platform, and with the instrument responses to the generic GICD (DRD - 2 1).

Both documents have been elaborated by MATRA MARCONI SPACE along with DORNIER and MMS Space Systems, Ltd.

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1. GENERAL

1.1. PURPOSE OF THE DOCUMENT

The General Instrument Interface Control Document (GICD) defines the requirements on the interfaces, tests and programme to which all the METOP **payload** complement instruments shall comply.

This document, along with the Instrument Interface Control Document (ICD) Outlines, will form the unique formal definition of interfaces for the METOP programme. Roth documents will then be configuration controlled by the METOP project team and formally signed off by the corresponding responsible authorities (TBD). In cases of conflict between the GICD and the ICD Outlines, the agreement or definition in the ICD Outlines shall govern.

The objective of these documents is to ensure that :

- instruments are designed, built and verified within the constraints imposed by the overall payload complement, platform and launch vehicle,
- the platform Prime Contractor is able to design, build and verify the platform in such a manner that the instruments can be successfully integrated into the system
- the spacecraft system can be successfully launched and operated to achieve the mission objectives of the METOP programme.

Note that this document is a preliminary specification that corresponds to the current definition status of the system (Phase A).

The requirements related to the electrical interfaces are dealt with in a dedicated Technical Note (General Avionics Instrument Interface. Control Document - GAICD, Ref. ME-IS-DOR-DOR-PM-0001), that is presented in annex to this document. It is aimed at including its contents in the relevant sections of the GICD in the future METOP project phase.

1.2. OVERALL METOP PROGRAMME

The METOP satellite is an element of the EPS/METOP system, that will be jointly developed by ESA and EUMETSAT. This system mission objectives are operational meteorology and climate monitoring from polar orbit, in order to complement the NOAA Polar Orbiting Environmental Satellite System.

The METOP satellite is composed of a platform (or spacecraft) and a set of instruments constituting the payload. This comprises :

Operational Meteorological Package

* Advanced Very-High Resolution Radiometer	AVHRR/3
* High Resolution Infra-Red Sounder	HIRS/3
* Advanced Microwave Sounding Unit - A	AMSU-A 1/A2

* Microwave Humidity Sounder	MHS
* Data Collection System	DCS/2
* Infrared Atmospheric Sounding Interferometer	IASI

- Climate Monitoring Payload

* Advanced Scatterometer	ASCAT
* Multi-frequency Imaging Microwave Radiometer	MIMR
* Scanner for Radiation Budget	SCARAB
* Global Ozone Monitoring Experiment	GOME

Note that the METOP programme comprises a series of two satellites : the first one is scheduled for an ARIANE 4 launch in late 2000, and the second one will be launched in line with the operational needs.

1.3. APPLICABLE DOCUMENTATION

AD1 ME-RS-ESA-SY-0001 Issue 1 (June 93) Space Segment Requirement Specification

2. MECHANICAL INTERFACE REQUIREMENTS

Each Instrument Interface Control Document (ICD) shall define the Mechanical Interfaces in accordance with the requirements specified below.

All the mechanical interfaces shall be defined and implemented using the international system of units (metric, SI).

2.0. SPACECRAFT MAIN CHARACTERISTICS

2.0.1. Axis Definitions

Spacecraft Absolute Reference Frame (0, X,, Y,, Z_s) F,.

This spacecraft-fixed coordinate system is used to define hardware location within the spacecraft. It is ideally defined as follows (Figure 2.0/1) :

- 0 is located within the spacecraft to launcher separation plane, at the centre of the attachment ring,
- The Xs-axis is perpendicular to this separation plane and oriented from the spacecraft towards the launch vehicle,
- The Zs-axis is the normal-out of the surface that carries the stowed solar array,
- The Ys-axis completes the right-handed orthogonal reference frame.

Local Orbital Reference Frame (G, X,, Y,, Z_{Lo}) F_{Lo}

The origin of the local orbital reference frame is the spacecraft centre of mass in the operational condition. The coordinates are known as the conventional pitch, roll and yaw systems. The yaw (Z_{Lo}) axis is directed towards the zenith, the pitch (X_{Lo}) axis is directed towards the orbit positive normal, and the roll (Y_{Lo}) axis completes the system : it is then along the anti-velocity vector (Figure 2.0/2).

Spacecraft Centre of Gravity Reference Frame (G, X,, Y,, Z_G) F_G

This frame is obtained by a simple translation of the spacecraft absolute reference frame F_S to the satellite centre of gravity (G). It is the reference for the mass, centring and moments of inertia configuration.

Attitude Reference Frame (X_{AOCS}, Y_{AOCS}, Z_{AOCS}) F_{AOCS}

This frame is obtained by the translation of the local orbital reference frame F_{Lo} to the position of the AOCS optical sensor (normal projection of the Earth sensor centre of mass on its mounting plane).

Spacecraft Optical Reference Frame (X_{SO}, Y_{SO}, Z_{SO}) F_{SO}

The Spacecraft Optical Reference Frame is defined by a master reference mirror cube (MRC) located at a stable position on the spacecraft. The unit vectors along X_{SO}, Y_{SO} and Z_{SO} will be nominally parallel to and in the same direction as the unit vectors along X_{AOCS}, Y_{AOCS} and Z_{AOCS}. The real orientation of F_{SO} unit vectors are defined by the normals of the mirror cube.

Instrument Interface Reference Frame (X_u, Y_u, Z_u) F_u

For each payload instrument, an interface reference frame shall be used for any alignment, or alignment stability requirements. It is defined from the mounting plane on the instrument side on a case by case basis, that shall be clearly **documented in the** ICD **Outlines. The axes are** preferably parallel to X_{SO}, Y_{SO}, Z_{SO} (Cf. § **2.1.2).**

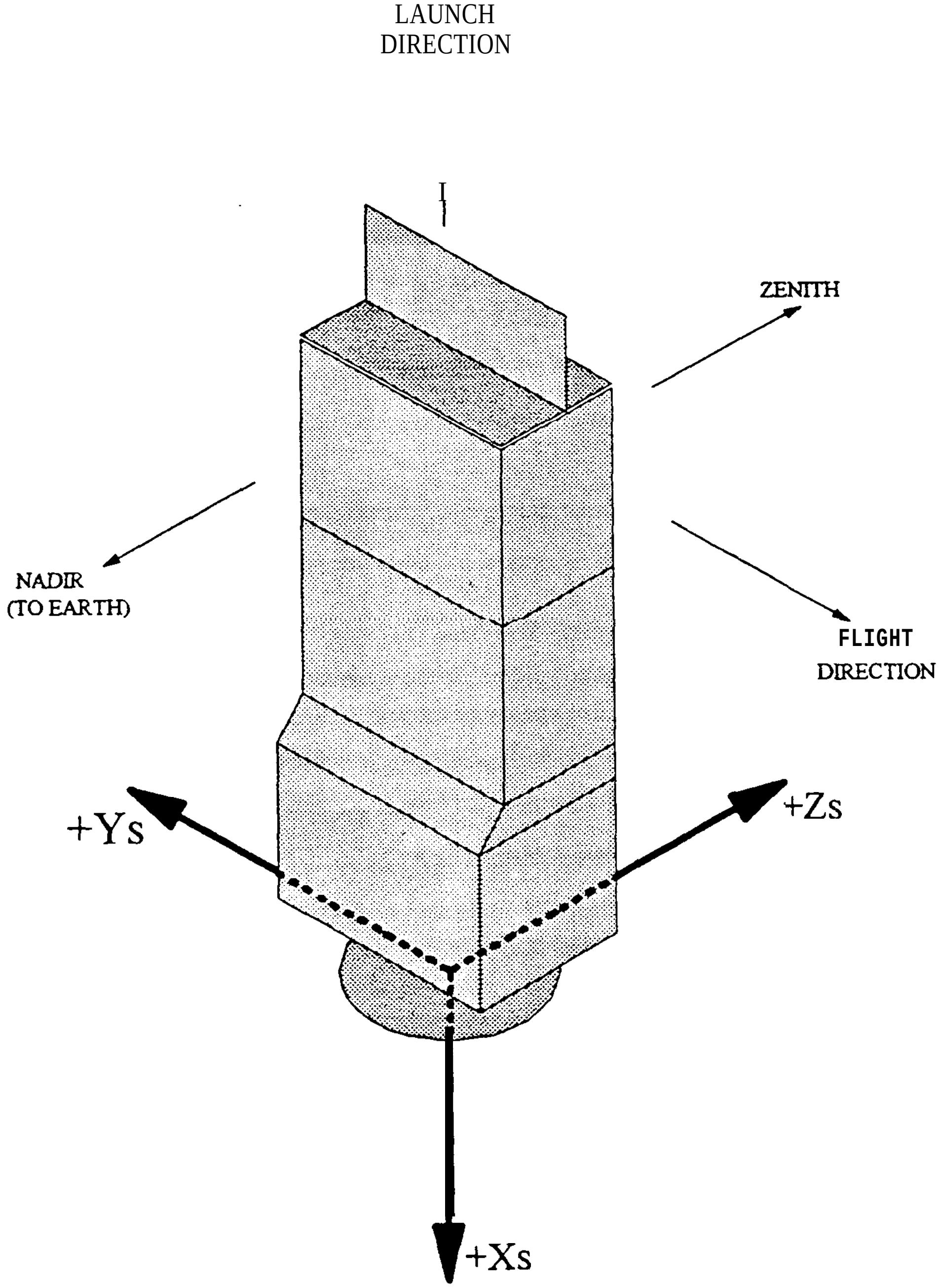

Figure 2.0/1 : Satellite Reference Frame

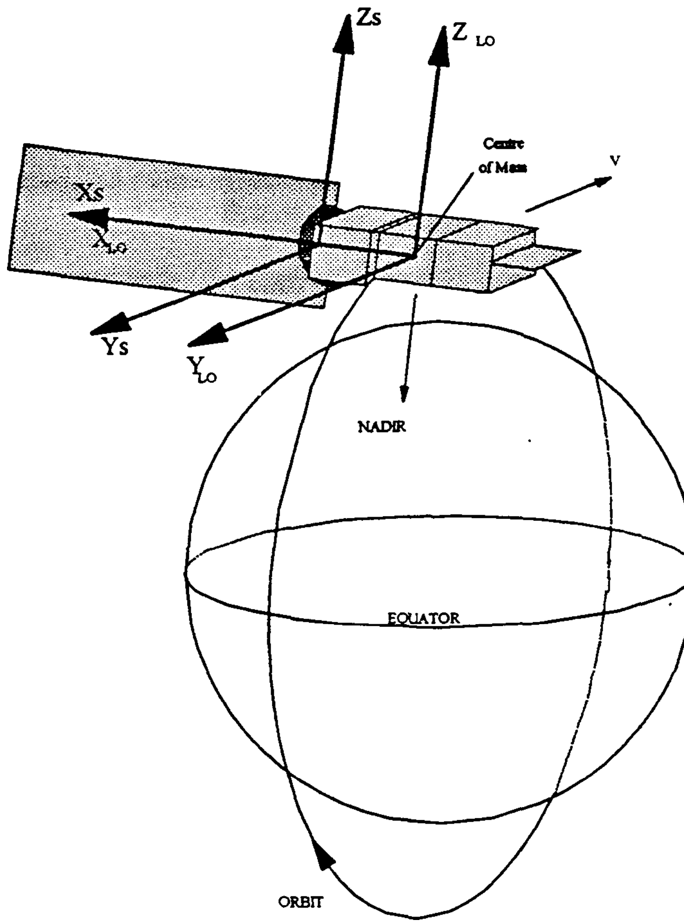

Figure 2.0/2 : Spacecraft and Local Orbital Reference Frames

2.0.2. Spacecraft Architecture Concept

METOP is a three-axis stabilised spacecraft that is built around a primary structure consisting of :

- a service module (SVM), which provides all standard service elements
- a upper payload module (PLM) that accommodates the different instruments and corresponding electronic equipments.

The service module is a box-shaped structure, that interfaces with the launch vehicle at the bottom and with the payload module at the top.

The payload module provides the main supporting structure and external panels on which are mounted the payload instruments. It also provides internal accommodation for both the payload support systems and the instrument electronic units.

2.1. INSTRUMENT PHYSICAL CHARACTERISTICS

Instruments shall be designed to be geometrically and mechanically independent from the platform in order to simplify and permit flexibility in the payload module configuration and integration activities. The instrument shall be designed to get either a single unit (containing all electronics, antennae, optical devices...) which is externally mounted onto the platform, and so finally integrated, either a set of different elements (electronics, antennae, harness, wave guides..) which are integrated at payload module level.

2.1.1. Module/Unit Identification

Each separately identifiable sub-assembly shall carry an identification with at least the following information :

- Equipment/Assembly Name
- Identification Part Number
- Serial Number

The location and method of marking shall be identified in the ICD.

The external unit dimensions in both launch and in-orbit modes, including mounting lug and connector envelopes, shall be recorded in the ICD. Unit overall dimensions shall be to a tolerance of 1 mm or 10% of the real size.

2.1.2. Mechanical interface Control Drawing

The instrument configuration and its interface requirements and dimensions shall be fully detailed in one (or more) Interface Control Drawing(s), that will be fully referenced by the Instrument ICD.

This drawing shall detail all coordinate systems utilised and their relationship to each other, together with the principal instrument interfaces.

The instrument shall have a right handed orthogonal co-ordinate reference system (X_u, Y,, Z,) F_u and it shall be defined such that :

- the origin shall be physically located on an accessible, identifiable instrument exterior feature (e.g. the centre of one mounting hole)
- the axes being ideally aligned with the X_s, Y_s, Z_s spacecraft axes (Cf. 2.0.1), e.g. for instruments mounted on the platform nadir side, the datum plane which shall contain the X_u, Y_u axes, is the plane containing the unit mounting lugs, and the Z_u axis is perpendicular to this datum plane in the direction from the unit to the datum plane.

These axes shall be referred to on all drawings and any finite element description.

The instrument physical characteristics shall be detailed together with indication of volumes required for moving and deployable parts. Where access to clear fields of view is required this shall also be shown.

Provision of instrument CAD models will be asked with the following media and file formats :

- operating system : HP-UX
- media type : 4 mm DAT tape
- media format : TAR
- file format (by order of preference) :
    - 3D IGES (assembly as one entity)
    - 3D IGES (assembly with each part in separate IGES file)
    - 3D IGES
    - DXF
    - 2D IGES

For the unit equipped with an optical reference cube, the normal to the faces of that cube will define the Unit Optical Reference Frame. This frame shall be nominally parallel to F_u.

Location of the mirror shall be clearly identified in the ICD drawing(s).

2.1.3. Mass Properties

The mass allocated to the instrument shall include the total instrument hardware that is intended to flight, i.e. instrument, mounting attachments and specific interface hardware if required.

Mass properties shall be established and recorded in the ICD. The record shall account for all mass states and mass dynamics attributable to deployable, consumable, moving or jettisonable materials or assemblies.

Mass shall be determined to the greater accuracy of 0.05 kg or 1% of instrument mass.

The Centre of Mass shall be determined to an accuracy better than 5 mm spherical error, in launch / deployed configurations, referenced to the instrument coordinate axes.

Moments of Inertia at the centre of mass location, about each major axis are to be determined for all instrument configurations to an accuracy of 1% of the total instrument moment of inertia for that axis, referenced to the instrument coordinate axes.

2.1.4. Instrument Induced Disturbances

Dynamic forces and torques induced by instrument operation that are reacted at the instrument to spacecraft mount shall be described and recorded, and comply with the limits specified here in. In case an instrument is composed of more than one independently mounted assembly, these requirements apply to each independently mounted assembly.

2.1.4.1. Non Recurring Transient Events

Forces and torques associated with, and the total momentum imparted to the spacecraft by instrument non-recurring events (e.g. during the release phase) shall be determined for each axis and their values recorded in the specific Instrument ICD outline, from the start to the end of the instrument internal motion.

This information shall be provided using the following standards :

- an analytical model including for example the rotation and/or the translation axis, the masses and/or the inertia in motion, the kinematic profiles, the actuators defaults and the static/dynamic unbalances of the moving part. All of these data shall be expressed in the instrument reference frame defined in § 2.1.2
- time measurements or simulation outputs of the forces and torques generated on each axis (expressed in the instrument reference frame defined in § 2.1.2)

2.1.4.2. Continuous and Recurring Transient Events

Forces and torques associated with, and the total momentum imparted to the spacecraft by instrument continuous and/or recurring events shall be determined for each axis and their values recorded in the specific instrument ICD Outline, from the start to the end of the instrument internal motion.

This information shall be provided using the following standards :

- an analytical model including for example the rotation and/or the translation axis, the masses and/or the inertia in motion, the kinematic profiles, the actuators defaults and the static/dynamic unbalances of the moving part. All of these data shall be expressed in the instrument reference frame defined in § 2.1.2
- time measurements or simulation outputs of the forces/torques/momentum generated on each axis, provided on a numeric tape, with the following characteristics :
  - sampling frequency greater than 1000 Hz
  - output format compatible with FORTRAN (double precision if possible)
  - measurement point defined and instrument reference frame used.

In both cases drawings presenting the corresponding time plots shall be provided with a time scale between typically 10 msec/cm and 1 msec/cm.

In addition, the associated Fast Fourier Transform (FFT) of these outputs should be provided with the normalization factor in both linear and logarithmic plots. The frequency content should be given between 0 and 100 Hz.

2.1.43. Induced Disturbance Torque Effect

The forces and torques which are computed at the instrument level, are translated to forces and torques expressed at the spacecraft centre of mass level, by considering the lever arms expressed in Table 2.1.4/1 (distances between the instruments reference points and METOP centre of mass).

INSTRUMENT	Xs	Ys	Zs	DISTANCE
AVHRR/3	- 4.7 m	- 1.3 m	+ 0.1 m	4.9 m
HIRS/3	- 4.7 m	- 0.5 m	+ 0.1 m	4.8 m
AMSU-A1	- 3.8 m	- 0.3 m	- 0.3 m	3.9 m
AMSU-A2	- 1.8 m	- 0.8 m	- 0.9 m	2.2 m
MHS	- 2.6 m	- 1.2 m	- 0.9 m	3.0 m
IASI	- 3.8 m	+ 0.4 m	+ 0.3 m	3.9 m
ASCAT ANTRF*	- 0.3 m	+ 1.4 m	-1.2 m	1.9 m
ASCAT ANTRA*	+ 0.1 m	+ 1.6 m	- 0.2 m	1.7 m
MIMR	- 3.2 m	- 1.0 m	+ 0.3 m	3.4 m
SCARAB	- 3.2 m	+ 0.5 m	- 0.9 m	3.4 m
GOME	- 0.2 m	- 1.3 m	- 0.8 m	1.6 m

* ASCAT : Distance between the deployable boom attachment points on the platform and the spacecraft CoG

Table 2.1.4/1 : Level arms between the instrument reference points and the spacecraft centre of mass (Values TBC)

Then the frequency templates of the AOCS transfer functions are used to evaluate the effects of the disturbing torque expressed at the spacecraft centre of mass on the spacecraft reaction wheels and on the spacecraft dynamics. The following templates shall be used :

- Figure TBD : $G_i(f)$ transfer function (METOP rate) / (disturbing torque) for i axis (three figures).
- Figure TBD : $H_i(f)$ transfer function (METOP wheel torque) / (disturbing torque) for i axis (three figures).

The impacts of a disturbing torque acting at the METOP centre of mass which is characterized by a bilateral Power Spectral Density $C_i(f)$ for i axis (1 σ value), can then be evaluated to the following rules:

- Inducedpointingerror: $$3\sqrt{\int_{-\infty}^{+\infty}\frac{|G_i(f)|^2}{(2\pi f)^2}C_i(f)\,\mathrm{d}f} \le \theta$$
- induced rate error : $$3\sqrt{\int_{-\infty}^{+\infty}|G_i(f)|^2 C_i(f)\,\mathrm{d}f} \le \dot{\theta}$$
- Induced torque error (reaction wheel) : $$3\sqrt{\int_{-\infty}^{+\infty}|H_i(f)|^2 C_i(f)\,\mathrm{d}f} \le C_W$$
- Kinetic Momentum : TBD

The maximum allowable errors, for any of the payload complement item, are defined in table TBD.

Instrument	X	Y	Z
θ (deg)			
$\dot{\theta}$ (deg/sec)		TBD	
Cw (Nm)			

Table TBD : Instrument Maximum **Allowable** *Errors*

2.1.4.4. Flexible Modes

TBD

2.15. Field of View Definition

Where relevant, the instrument shall define in its specific ICD outline its field of view requirements referenced to the unit coordinate system.

The definition shall cover the following points :

scanning mode presentation and scanning plane definition ;

instrument vertex as a three dimensional position in the unit system, and aperture shape

instrument boresight direction, as the centreline or nadir line of the instrument field of view (if possible)

instrument instantaneous field of view (IFOV)

instrument field of view : its definition shall consider different types of limitations for different zones of the field of view, clearly justifying the technical bases for the different limitations. For instance, in a first zone, the requirement could be the absence of other bodies, in a second zone, the absence of highly reflective bodies, and in a third zone, the absence of specular reflecting surfaces. These fields of view shall be defined in angles related to the boresight direction.

In addition, the instruments shall define any requirement on solar exclusion field of view.

2.2. INSTRUMENT MOUNTING ATTACHMENTS

The attachment points shall be designed to guarantee the connection of the instruments to the platform structure throughout the ground and orbital life of the spacecraft.

23.1. Method

Instrument mounting shall be described in the ICD and should be accomplished by bolts, passing through instrument flanges, lugs or structural components, which mate with spacecraft supplied hardware. At least four bolts shall be used to attach each independently mounted instrument assembly to the spacecraft. Instrument mounting bolts shall be ISO-metric fine thread titanium bolts used with steel alloy washers. Instrument designs shall incorporate mounting provisions of sufficient size, number and location to assure survival of the worst combination ‘of ultimate level conditions, including thermal differential loadings, without permanent deformation or damage to either instrument or spacecraft.

Instrument assemblies to be mounted to the SVM shall be designed to use M4 bolts for attachment. A minimum spacing between these attachments shall be 25 mm.

Instrument assemblies intended for external mounting on the PLM shall be designed to use M5 or M6 (M5 preferred) bolts for attachment. Minimum distance between attachments shall be 100 mm.

Mounting bolt size, number and location shall be recorded in each instrument ICD. Bolts shall be supplied by the instrument contractor.

23.2. Reference Point (Hole)

Mounting compatibility with the spacecraft will be assured by the use of matched precision drill templates provided by the instrument supplier. The location of all Reference points shall be fully dimensioned in the interface control drawing.

All attachment holes shall be located with respect to a master reference point with tolerances no greater than the following :

- Distance: ± 0.1 mm
- Pitch Circle: Radial ± 0.1 mm

  Angle ± 1 arc minute

2.23. Mounting Surfaces

The spacecraft will provide a planar mounting surface to which the instrument is bolted. The surface shall be flat to less than 0.1 mm in 100 mm and have a surface roughness of 1.6 micron RA. The mounting surface shall be parallel to one of the spacecraft principal reference planes. The Instrument mounting attachments shall be compatible with this interface.

The instrument mounting surface characteristics (including planarity) shall be recorded in each instrument ICD.

2.2.4. Materials

Materials used shall be selected from the ESA preferred materials list (PSS-01-701), or shall be demonstrated to conform with the requirements of PSS-01-701, applicable documents.

Where considered applicable, the effects of Atomic Oxygen with possible synergistic effects from thermal cycling and near/far ultra-violet, shall be addressed.

All metals shall be either corrosion resistant, or suitably treated using an ESA approved process to resist corrosion (including stress corrosion).

Where it can be avoided, dissimilar metals shall not be used in intimate contact with each other unless treated by an approved process to resist electrolytic corrosion.

A full material and processes list shall be contained in the Instrument Interface Control Document.

235. Interface Loads

The then-ml coefficient of expansion of the platform mounting surface is 2.0 x 10^{-6} PC (CFRP) for the purpose of determining instrument attachment loads and designing instrument provisions. Some instrument may not withstand such an interface. This shall be negotiated on a case by case basis, and the design of a specific interface hardware will then be considered.

Spacecraft instrument-attaching hardware (e.g. threaded inserts) shall accommodate axial, radial and torsional loads in accordance with Table 2.2/1 without yield or failure.

From these, instrument attachment load limits shall comply with the following :

$$(T/T_m)^2 + (S/S_m)^2 + (M/M_m)^2 \leq 1.0$$

where,

- T, S and M are the actual values of Tension, Shear and Moment loads applied to the attachment bolt.
- T,, S_m and M_m are the insert strength capability specified in Table 2.2-l.

Note: Compression (C, C_m) where capability is specified, shall be used instead of tension (T, T,) if a lower instrument load limit would result.

Compliance with these load requirements shall be shown in the ICD.

METOP PANEL	TENSION (N)	COMPRESSION (N)	MOMENT (N.mm)	SHEAR (N)
± Y	4770	4950	43313	3816
+ Z	4770	4950	43313	6311
- z	4770	4950	43313	5064

Table** 2.2-1: **Spacecraft Insert Strength for Instrument Mounts

2.2.6. Accessibility

Instruments shall not require assembly or disassembly to effect mounting on or dismounting from the spacecraft.

The instrument design shall permit easy access to mounting bolts and to test points and components (i.e. oscillators) that may require adjustment.

Interface Drawings and descriptions shall identify these points.

22.7. Grounding Point

Titanium and **Ti Alloy** bolts used for grounding on Aluminium structures, shall be used with stainless steel washers between the head of the bolt and the Aluminium structure.

All instruments shall be capable of being grounded via a single point attachment and this shall be clearly identified.

The physical definition of the ground stud is given in § 5.

23. POINTING REQUIREMENTS

The instrument shall define in the specific ICD its required pointing performances at the instrument interface reference frame. They shall be expressed in 3σ values for any of the three axes for a specific frequency bandwidth (typ. O-4 Hz), and cover the following characteristics :

- Absolute pointing error (separation between the actual and comma&d pointing directions)
- Absolute measurement error (separation between the actual and on-board measured pointing directions)
- Absolute rate error (separation between the actual and the commanded pointing rate).

The platform pointing performances are presented in § 10.2.2.

2.4. ALIGNMENT

The integration / alignment of all instruments on the METOP platform is a Prime Contractor responsibility. It is the instrument responsibility to provide an instrument / unit design which is compatible with the required alignment accuracy.

2.4.1. Optical Reference Cube

Instruments requiring precision alignment shall be equipped with an optically reflective cube. This unit optical reference frame will be accurately aligned on-ground with respect to the Spacecraft Optical Reference Frame $(X_{SO}, Y_{SO}, Z_{SO})\ F_{SO}$. The instrument cube shah have on each reference face a surface of at least 645 sq. mm. Reference faces shall be orthogonal to each other within ± 1 arc second.

During the alignment phase, the unit optical reference frame will be aligned with respect to the spacecraft optical axes. Thus special attention shall be paid to the criticality of this reference, and therefore on the reliability of its fastening system and the possibility of providing a redundant reference.

The reference cube(s) shall be permanently affixed to the instrument. The cube(s) shall be located on the instrument such that two faces are accessible for direct viewing by an external (to the spacecraft) mounted theodolite. Viewing will be realised when the instrument is mounted on the spacecraft, without the removal of other spacecraft instruments, subsystems or hardware items.

Optical cube installation location, orientation and direct viewing access shall be documented.

2.43. Alignment Procedure

Each instrument shall be equipped with suitable means (i.e. shims, screws, eccentrices...) to adjust the unit alignment on the platform and these means are part of the instrument deliverables. When shims are provided, their minimum resolution shall be equal to 3/100 of mm.

Resolution of these adjustment means : TBD

For each instrument, a description of the adjustment method shah be provided for approval. This shall include as a minimum a definition of the instrument adjustment range (e.g. minimum and maximum tilt angle) and a detailed description of the hardware used for that purpose.

At last, the instrument designer shall demonstrate that the adjustment activities will not introduce stresses in the instrument and in the platform structure (or that the stresses are quantified and stay below an acceptable level).

2.43. Co-Alignment

General establishment of instrument co-alignment shall be by dimensional control of the instrument mounting hole locations with respect to the critical axes of the instrument. This is related to locations of mating spacecraft hardware that is dimensionally controlled with respect to the spacecraft reference frame.

Co-alignment requirements between instrument interface feet shall be documented in the instrument ICD. The following co-aligmnent will be realised:

- AMSU A1/A2 0.05 deg (3σ) with respect to AVHRR/3
- HIRS/3 0.05 deg (3σ) with respect to AVHRR/3
- IASI, 0.05 deg (3σ) with respect to AVHRRA
- MHS 0.05 deg (3σ) with respect to AVHRR/3

25. STRUCTURAL DESIGN

25.0. General Requirements : Load Cases

The instrument shall be designed to withstand exposure to the environments it will encounter during its life time without degradation to its performances and without detrimental influence on the spacecraft of other instrument performances.

The following load types shall be analysed concerning their applicability, magnitude, time/duration of occurrence for each structure :

- quasi-static accelerations
- low frequency transient accelerations
- high frequency random accelerations
- coupled analysis loads
- sine vibration loads.
- shocks (e.g. separation, deployment of other instruments, self-induced...)
- acoustic noise
- differential pressures (venting)
- thermal gradients
- loads induced by different thermal expansion coefficients of mated materials
- assembly forces (e-g.: torqued bolts, marmon clamps)
- thrust of propulsion systems
- loads induced by mechanical operation
- latching/unlatching forces/torques
- fluid dynamic effects
- residual stresses
- loads due to handling(assembly, integration, test) and transportation

The principal cases judged relevant to an instrument &sign shall be identified.

In order to assure the structure will withstand without failure all experienced loads and environments, the structural designer shah derive and document in a single report dimensioning load cases from the critical combinations of load types and environments identified during all phases of structural life.

The life phases shall be investigated, and described, systematically from manufacturing, storage, transport, handling, testing, refurbishment, launch, ascent and in-orbit events.

All instrument loads shah be reacted at the instrument mounts. Static and dynamic loads may be applied to the instrument in any direction.

The instrument shall be capable of operating in both a one gravity load (applied in any orientation) and a zero gravity load environment.

25.1. Limit Loads

The instrument design shall use "A" value strength allowables and other physical properties from :

- MIL-HDBK-5 for metals

- MIL-HDBK-17 for laminates and bonded structures.
- CFRP stiffness properties shall use mean values.

Design limit loads shall be defined as the maximum anticipated load, or combination of loads, which the structure is expected to experience during its lifetime. The effects of environmental phenomena acting at the time of the design condition must be included (e.g.: elevated temperatures).

2.52. Quasi-Static Design Levels

For METOP preliminary design purposes, the quasi-static design load factors derived from Figure 2.5/1 shall be applied. It shall be understood that Figure 2.5/1 is only a design tool, it does not imply that a static test must be performed to the levels indicated by it.

These factors have to be applied for determining loads and their use :
- applied at the instrument Centre of Mass.
- into the principal instrument/spacecraft axis giving the worst case reactions
- it should be taken that the loads are not acting simultaneously

For all instrument with masses larger than 100 kg the factor is constant and equal to 15 g.

Detailed load factors for design and test purposes for the instrument individual structures will be issued after the coupled analysis of the complete spacecraft.

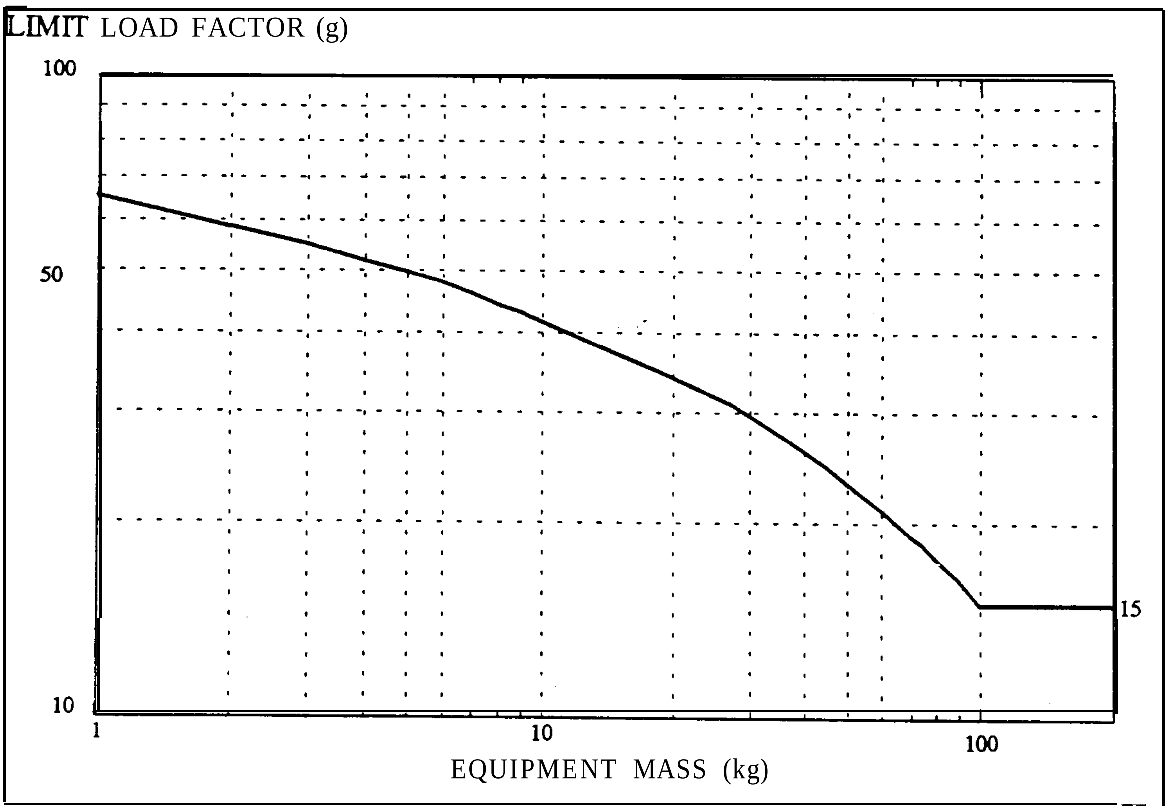

Figure 2.5/1 : Quasi-Static Design Load Factors

2.5.3. Safety Factors on Design Limit Loads (Factor of Safety)

A factor of safety is a multiplying factor greater than, or equal to, unity. This is applied to a design limit load and accounts for uncertainties in the definition of design loads, analytical simplifications, manufacturing tolerances... Table 2.5/2 reflects the factors to be used for dimensioning a mechanical item. With the exception of proof factor, they are applicable to design limit loads for evaluation of the Margin of Safety.

It is recalled that the Margin of Safety is defined as the following ratio :

$$M.S.=\frac{\sigma_f}{j.\sigma}-1>0$$

with σ_f : Material characteristics at failure point (allowables from material data sheet)

σ : Design limit load

j : Relevant safety factor

The Design Limit Loads applicable to a particular instrument are TBD.

Test Levels shall be :

- Qualification tests are 1.25 x Flight Limit Loads
- Proof testing shall be to 1.2 x Flight limit loads
- Acceptance tests are 1.1 x Flight Limit Loads

Local loading (inserts) shall have an Ultimate safety factor of 2.0 except for payload attachment inserts which shall have an ultimate safety factor of 15.

2.5.4. Dynamic Characteristics and Structural Mathematical Models

2.5.4.1. Stowed

As a basis all instrument units shall show a first natural frequency higher than 100 Hz when bolted at their flight interfaces to a rigid fixture. Resonances of internal items (PCB's, discrete large components) should also be higher than 100 Hz to avoid coupling with inputs from the launcher / Spacecraft, which as a consequence of high amplification factors might compromise their functional capabilities. As soon as this requirement (> 100 Hz) is met, no mechanical interface model will be required.

Exception to the above requirement is admitted for large / heavy instrument units (to be determined on a case by case basis). These units shall then show a first natural frequency higher than 60 Hz when bolted at their flight interfaces to a rigid fixture. This second class of instruments shall predict early in their development phase, their dynamic characteristics utilising a finite element model. Hence, the Instrument Contractor shall provide the following items :

- a validated Reduced Interface Finite Element Model in NASTRAN format that will be integrated into the overall structural mathematical model of the Satellite. The Reduced Finite Element Interface model shall have approximately 150 degrees of freedom.

Non-Pressure Load Cases	MINIMUM SAFETY FACTORS			
	Yield	**Ultimate**	**Proof**	Buckling
General Structure - Metallic				
Verified by ***Analysis*** **and Test**	**1.1**	**1.5**		**1.5**
Verified by Analysis only	1.25	2.0		2.0
General Structure - Non-Metallic				
Verified by Analysis and Test		1.5	1.2	1.5
Verified **by Analysis only**		3.0		3.0

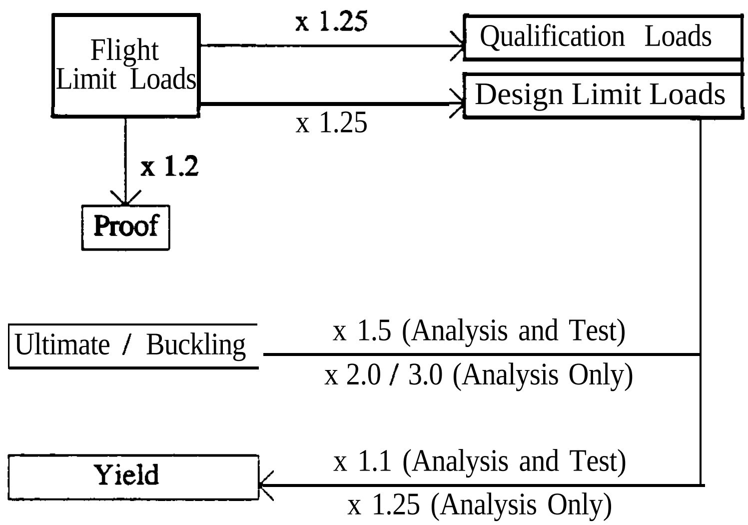

Figure 2.5/2 : Safety Factors

- a stiffness representative Structural Model of the instrument for integration on the Satellite Structural Model for satellite level vibration testing.

The above models shah be supplied with all the relevant documentation to understand and run the model.

Note that the above requirements apply to all hardware and shall be demonstrated by analysis and confirmed by test.

For instruments whose first natural frequency has been derived solely by analysis, the above frequencies of 60 and 100 Hz, shah be increased by 10% to 66 and 110 Hz.

2.5.4.2. **Deployed**

The first natural frequency of the METOP instrument in its deployed configuration shall be above 3.5 Hz. The provision of a mechanical model (Cf. § 2.5.4.1.) is required.

2.6. MECHANISMS

2.6.1. Functional Requirements

Instruments employing electromechanical devices shah be functionally analysed to determine loads deriving from their activation, both in orbit or on ground, as applicable. These mechanisms shall be designed to minimize static and dynamic disturbances to the platform.

Some mechanisms, to be agreed on a case by case basis (i.e. aperture door mechanism), shah have devices providing position tell-back. When release by one shot actuators is proposed, they shah be capable of being operated mechanically for test purpose. They shah allow re-installation of new actuators without need for major disassembly of the mechanisms, when integrated on the platform.

2.6.2. Performance Requirements

Mechanisms shah be designed to the same criteria as all other structural items. They shah therefore withstand without degradation all the environments they will be subjected to during then life. They shah perform within their specifications during the complete lifetime, and shall be compliant with the EMC requirements.

Electrical motors shah be able to withstand their corresponding stalling torque during at least 1 minute without any permanent damage.

The mechanism shall be sized for any extreme values of the corresponding components of resistance :

- Inertia	multiplied of divided by	1.1
- Friction	multiplied of divided by	3.0
- Hysterisis	multiplied of divided by	3.0
- stiffness	multiplied of divided by	1.2

As a minimum, two extreme cases shah be envisaged :

- the maximum predicted action torques / forces against the minimum reaction torques / forces

- the minimum predicted action torques / forces against the maximum reaction torques / forces

The minimum action torque / force against 2 times the predicted or measured worst case reaction torques / forces induced by friction, hysterisis or stiffness, shall be able to initiate the motion whatever the starting point (the components of resistance shall be multiplied the same factors as above).

The mechanisms shall be designed for the maximum number of the cycles expected during the mechanism life. The minimum required functional safety factors on the number of cycles is 4. Life cycle testing shall consist in performing 4 times the nominal number of cycles or the nominal number plus 10 cycles, whichever is greater.

Deployable Antennae or Booms

Separate tests (TBD) shall be performed to demonstrate the operation of deployment under the spacecraft loading conditions and the withstanding of pyrotechnic release shock.

Note that the deployable antennae or booms shah withstand in their deployed configuration the spacecraft spin loading conditions that occur in Safe Mode, without degrading the pointing performances of the deployed item.

2.7. PYROS

TBD

2.8. INSTRUMENT APERTURE COVERS

2.8.1. Sensor Covers

The units that incorporate a sensor aperture shall be delivered with a dust/protective cover over the aperture. These covers can either be removable before flight or part of the instrument and so deployed/retracted in orbit.

2.83. Removable Covers (Non-Flight Items)

All removable covers shall be :

- normally removed during system test when flight configuration is mandatory, i.e. thermal vacuum testing or vibration testing unless other factors have priority (e.g. contamination prevention) ;
- accompanied by a detailed procedure for their removal during integration ;
- easily identifiable as non-flight hardware. They shah also be clearly marked as non-flight items on the relevant drawings.

2.83. Deployable Covers (Flight Items)

Any unit that absolutely requires an aperture cover at all stages of spacecraft integration and launch shall incorporate a deployable cover activated by telecommand.

This deployable cover shall be classified as a **mechanism** and the deploymentketraction mechanism **shall meet the** design **requirements** of $2.6. The variations of **centre** of gravity and Moment of Inertia of the **instrument produced by the** deployment shall b[redacted]ed.

The cover deployment and retention system shall be described in the corresponding instrument ICD.

The cover deployment system used shall take into **account** the fields of view and other requirements **from** adjacent **instruments** and/or units. This system and the way to operate it shall therefore be agreed **with** the **METOP** project team during the design phase. Solutions that **minimize** the variations of the required **instrument** volume during/after deployment will be preferred.

28.4. Purging **Interfaces**

The location for the purge valve shall be clearly defined.

3. THERMAL INTERFACE REQUIREMENTS

3.0. SPACECRAFT THERMAL CONTROL CONCEPT

The METOP spacecraft will consist of two separate modules : the Service Module (SVM) and the Payload Module (PLM). Each module has its own thermal design. Thermal decoupling between the modules will be satisfied by ensuring low conductivity at the interface and allowing therefore service module and payload module thermal design to proceed on an individual basis.

The thermal design of the METOP spacecraft will be an optimised passive design with controlled electrical heaters. As a general rule, the satellite thermal subsystem shall control the interface temperature of all the payload units.

3.1. INSTRUMENT THERMAL CONTROL CONCEPT

3.1.1. Category

An instrument or instrument unit shall be defined in the ICD as being one of the following categories :

Category A **:** Individually Controlled Units

Category A instruments/units shall be thermally isolated from the platform. They are conducfively and radiatively decoupled from the platform structure and perform their own thermal control. The thermal design of these units shall be under the responsibility of the instrument contractor, controlled by defined interfaces (heat exchange requirements, interface temperature specifications, thermal model) between the unit and the platform.

Category B : **Collectively Controlled** Units

Category B instruments/units shall be thermally integrated with and controlled by the platform thermal subsystem. Conductive coupling to the platform structure and radiative coupling to a platform controlled environment will be defined taking into account unit internal characteristics and conductive interface temperature specifications.

Category C

Category C instruments/units shall be those where the thermal control responsibility is shared between the instrument contractor and the platform thermal control subsystem.

As a baseline, all sensors belong to the A category, and all electronic units to the B category (Cf. Table 3.1/1).

3.12. Responsibility

The thermal control philosophy of each instrument, module or unit shall be defined in the instrument ICD. This shall include a statement of the major heat flows away from each item and how this is achieved. The instrument contractor is responsible for the provision of adequate information about the internal design, and the corresponding simplified thermal mathematical model.

Instrument	Category
AVHRR/3	C
HIRS/3	C
AMSU-A1	A
AMSU-A2	A
MHS	A
DCS/2	B
IASI	
Sensor	A
Electronics	A or B
ASCAT	
Antennae and SFE	A
Electronics	B
MIMR	
Sensor	A
Electronics	B (TBC)
SCARAB	A
GOME	A

Table 3.1/1 : Instrument Thermal Concept Category

Category A

The instrument contractor shall be totally responsible for the internal and external thermal &sign of the instrument, module or unit_ During normal operation the instrument thermal &sign shall control the temperature of the module within the required operating temperature range. The instrument contractor shall not rely on a heat flow to or from the platform for the thermal control of the module. The instrument contractor shall define and integrate heaters and thermostats which shall be used for the thermal control of the module during nonoperating, standby and lower modes of operation. These thermostatically controlled heaters will be powered directly from the platform thermal control units. **Both main** and redundant thermostatically controlled heater circuits shall be provided. The instrument contractor shall provide and integrate all thermal finishes and insulation blankets (MLI) on the module.

Category B

The instrument contractor shall be responsible for the internal thermal &sign of the unit_ The platform thermal control subsystem shall be responsible for maintaining the thermal environment of the unit within acceptable temperatures during all unit modes of operation. The platform thermal control subsystem will provide all necessary heaters thermistors and thermostats. The platform thermal control subsystem will define the external thermal finish required for the unit. The instrument contractor shall be responsible for the application of all paints and conversion coating required, tapes and other finishes will be supplied by the platform thermal control subsystem who may be responsible for their application if requested.

Category C

The instrument contractor shall be responsible for the internal and external thermal design of the instrument or unit but where thermal control measures are required by the platform thermal control subsystem, this shall be specified in the instrument ICD. The platform thermal control subsystem shall be responsible for all thermal control hardware external to the instrument/unit necessary to meet these requirements The instrument contractor shall define and integrate heaters and thermostats which shall be used for the thermal control of the instrument/unit during non-operating, standby and lower modes of operation. These thermostatically controlled heaters will be powered directly from the platform thermal control units. Both main and redundant thermostatically controlled heater circuits shall be provided. The instrument contractor shall provide and integrate all thermal finishes and insulation blankets (MLI) on the instrument/unit. The instrument contractor shall be responsible for the definition and integration of all thermistors necessary for the thermal monitoring of the instrument/unit.

Harness Thermal Control

TBD.

3.2. INSTRUMENT TEMPERATURE REQUIREMENTS AND THERMAL CONTROL BUDGETS

3.2.1. Temperature at Conductive Interface

The ICD shall define the following acceptance and qualification temperature requirements for each module or unit which comprises the instrument :

- Operating Temperature Range
  (independent of the instrument operating modes)
- Non-Operating Temperature Range (Survival),
  As a baseline, the instrument is not switched on when this temperature is applicable.
- Unit Switch-On Temperature
- Temperature Stability
  It will be maintained when the operating temperature range is applicable.

The Temperature Reference Point(s) (TRP) at which these temperatures apply shall be defined in the ICD. Note that the TRP will be monitored by the platform thermal control subsystem, and is on the spacecraft side of the interface. The platform temperature ranges at the temperature reference points for the different units are specified in Figure 3 .2.1/1.

	Platform Temperature Operating (deg. C)	Platform Temperature Non Operating (deg. C)	Platform Temperature Stability (deg. C)
Category A			
Category B		TBD	
Category C			

Figure 3.2.1/1 : Platform Temperature Ranges

Each contractor responsible for thermal &sign shall provide temperature specification at the conductive interface for thc item under their responsibility so that the contractor responsible for the other side of the interface can use these temperatures in their thermal design and analysis. These temperatures shall not, however, supersede any temperature requirement specified in a higher level specification.

322. Environmental Temperature (Radiative Interface)

The instrument contractor shall define in the ICD the desired radiator fields of view to space. This may be expressed as a geometric view factor or preferably as either a Gebhart radiative factor or radiation term (W/K^4), which &fines the total radiative view to space, both by direct view and by reflections from adjacent surfaces.

Any special interface requirements or definitions shall be specified in the ICD. Examples of this include the need to prevent solar illumination of parts of the instrument.

Instrument contractors responsible for the external thermal design of instruments, modules or units will be provided with an overall geometric model of the spacecraft and its instruments which shall be constructed by the spacecraft thermal control subsystem using the interface geometric models supplied by other instrument contractors. The spacecraft thermal control subsystem will provide temperatures for these surfaces and a sink temperature for each instrument.

3.23. Heater Power Budgets (Categories A and C)

The instrument contractor shall define in the ICD the resistance and tolerance of each heater circuit in the instrument, module or unit_ The heater power budgets for these heaters will be defined for the various modes of operation of the instrument. Note that, in nominal operations, the heater power shall remain within the total power allocation. It shall be stated whether this heat power is supplied on a dedicated power bus or not.

3.2.4. Instrument Thermal Dissipation

The instrument contractor shall define in the ICD the thermal dissipation of each instrument, module or unit during the various modes of operation of the instrument. Both peak and orbital average dissipations shall be defined_ For Category B units the instrument contractor shall define in the ICD any non uniform heat flux / dissipation at the unit baseplate which may affect the thermal control and temperature of the unit.

335. Heat Exchange Budgets

For Category A instruments, modules and units the maximum heat flow in or out of the platform shall be less than 5 Watts. Exceptions to this may be considered on a case by case basis. The instrument contractor shall supply the calculated heat flows between the instrument, module or unit and the platform, positive values being used for heat flow from the platform to the instrument.

For Category B units the instrument contractor shall supply the heat flows to or from the platform obtained from the unit thermal analysis. The conductive and radiative heat flows shall be specified separately, positive values being used for heat flow from the platform to the instrument.

32.5. Thermo-Elastic Interface

The &sign performances of the mechanical mounting attachments shall also take into account the thermal loads encountered during all mission phases. The thermal coefficient of expansion of the materials either side of the interface shall be defined in the ICD.

33. THERMAL INTERFACES

The general characteristics of the instrument to spacecraft thermal interface and the requirements for its thermal control shall be as given below. For each instrument, a detailed description of its interface and thermal requirements shall be included in the instrument ICD.

33.1. Thermal Interface Drawing

The instrument contractor shall prepare and supply Thermal Interface Drawings which shall define the total thermal interfaces. These drawings and their issue shall be included in the instrument ICD. The interface requirements given below may be defined either in the ICD or in this Thermal interface Drawing. It shall, at least, contain the following data :

- overall layout
- dimensions - ovetall size including thickness and their attachment
- Temperature Reference Point (TRP)
- radiator areas
- external surface optical properties
- apertures (position and size)
- blankets
- blanket performance
- optical properties of box in/outside and protruding parts, apertures... (BOL, EOL if applicable)
- non operational heater location
- spacecraft powered thermistor location (if applicable)
- grounding of MLI

33.2. Conductive Interfaces

The ICD shall contain a definition of the conductive **interface.** If this is the same as the mechanical interface then specific reference shall be made to the relevant section or drawing in the ICD.

For each Category B unit the baseplate contact area with the PLM shall be defined.

333. Radiative Interfaces

The external finishes of the instrument (MLI ,coatings, finishes etc.) shall be defined along with their optical properties at BOL and EOL.

For each instrument radiator, its area, field of view, required radiator temperatures and rejected heat shall be defined in the instrument ICD. The instrument thermal design shall be capable of some adjustment to the radiator size and finishes to obtain a satisfactory thermal design with the fields of view obtained with the spacecraft configuration.

33.4. Thermal Heat Capacity

The Thermal Heat Capacity of each unit or module comprising the instrument shall be defined in the ICD.

335. Instrument Temperature Measurement

The location, type and electrical interface of all devices used for instrument temperatnre measurement shall be defined in the ICD.

33.6. Thermal Mathematical Models and Analysis

Analysis

Conuactors responsible for thermal design shall demonstrate the thermal performance of items under their responsibility by analysis using well tested established software. For Category C items where the responsibility is shared, the instrument contractor shall be responsible for the initial thermal analysis of the external thermal &sign of the instruments, module or unit to establish the requirements on the spacecraft contractor to be defined in the ICD.

For METOP the following software packages may be used :

SINDA	ESATAN
THERMICA	MATVIF and MATFLUX
TRASYS	ESARAD

In order to ensure compatibility between various elements, to provide understanding between contractors, and to facilitate exchange of mathematical and geometric models, it is essential that there are restrictions on the features used that these software packages provide. Such restrictions shall be as defined in TBD.

Analysis Cases

The contractor responsible for the thermal design shall establish the analysis design cases to be analysed to demonstrate the acceptability of the thermal control design throughout the various mission phases. As a minimum, the following shall be considered :

- BOL and EOL
- Maximum and minimum environmental conditions
- Steady states and transient analyses
- stowed and deployed

Analysis Reports

Analysis reports which facilitate the requirements, understanding and definition of the thermal interface shall be delivered to the relevant contractors via the spacecraft contractor.

33.7. Thermal Interface Models

The instrument contractor shall provide interface models (in ESATAN) to the spacecraft contractor. The purpose of these interface models shall not be to predict instrument temperatures but to provide thermal interface data to be used by instrument contractors responsible for surrounding instruments in their thermal &sign and analysis. These interface models shall consist of mathematical models and geometric models which shall :

- enable both steady state and transient analyses to be performed
- have optical properties for both BOL and EOL analyses
- be representative of all interface heat fluxes
- where appropriate, enable different configurations of the instrument (e.g. stowed and deployed) to be analysed.

The full requirements for delivered mathematical and geometric models shall be as defined in TBD.

3.4. THERMAL ENVIRONMENTAL CONDITIONS

3.4.1. Mission Phases

The instruments thermal control design shall ensure that the instruments are maintained within acceptable temperature limits for the following mission phases :

- Pm-launch
- Launch
- First acquisition phase
- Nominal Operation
- Contingency modes (safe modes)
- Reacquisition phase

3.4.2. Attitude

The attitude of the METOP spacecraft will vary depending on the phase of the mission :

First Acquisition Phase

After separation from the launch vehicle the METOP spacecraft may have any attitude for a maximum time period of 60 minutes (TBC). During this period all payload deployable elements will be in a stowed condition and the spacecraft will be thruster controlled.

Nominal Operation

During this phase the METOP spacecraft will be Earth pointing with the spacecraft -Z axis pointing earthward along the Earth ellipsoidal local normal. The spacecraft -Y axis will be close to the velocity vector. The spacecraft Z axis (yaw) will be steered according to a sinusoidal function over the orbit with an amplitude of approximately 4 degrees (TBC).

Safe Mode Attitude

In this mock the spacecraft will point its principal inertia axis to the Sun within an error cone of ±15 deg. during sunlit phases. When leaving eclipse, the error cone angle may be as high as 30 deg. due to drift during eclipse. The error cone of ±15 deg. will be achieved within 300 sec. after the end of the eclipse period. During safe mode, the METOP spacecraft will rotate about its principal inertia Z axis with a rate of 0.5 ± 0.1 deg./sec. During safe mode the spacecraft will be thruster controlled.

Re-Acquisition

In the event of AOCS failure, the nominal operation attitude may be lost at any time during the mission The maximum duration of this loss of nominal operation attitude shah be 7000 seconds. A minimum rotation rate of TBD deg/sec about any axis shall be assumed during this time. The spacecraft is thruster controlled during this phase.

Attitude During Out-Of Plane Manoeuvres

In order to apply thrust in the out-of plane direction, the METOP spacecraft will be rotated about the spacecraft Z axis by ±90° with respect to the nominal attitude for a maximum duration of 2 periods of 30 minutes duration per orbit. The interval between such manoeuvres will be approximately 100 days.

3.43. Operational Orbit

The METOP spacecraft will operate in a Sun synchronous circular orbit with the following characteristics :

- Altitude 818 ± 30 km
- Descending node time: 9:00 am ± 5 minutes
- Orbit Inclination: 98.705 deg.

The variation of Solar Aspect Angle (90°-beta angle) throughout the year is given in Figure 3.4/1. The variation of the angle between the orbit plane and the Sun (beta angle) wit respect to the solar declination is given in Figure 3.4/2.

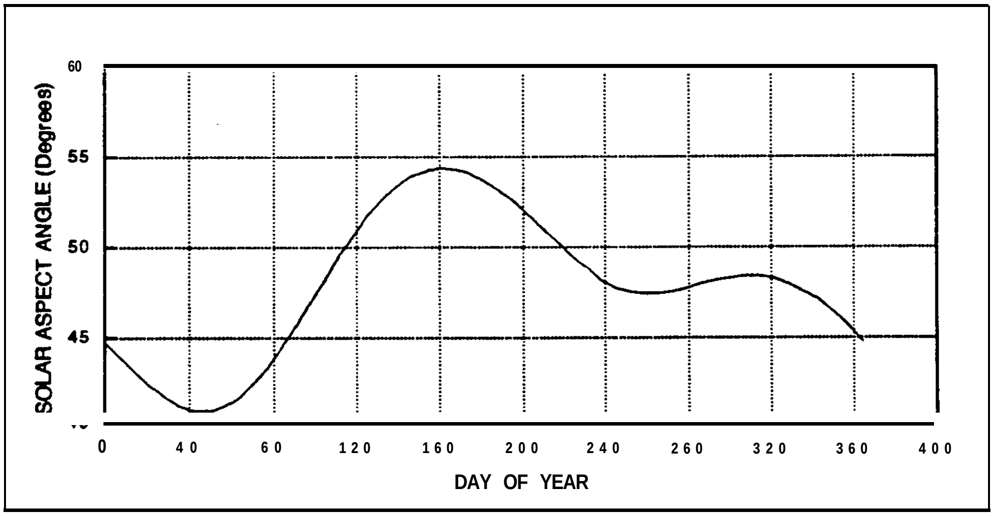

Figure 3.4/1 : Variation of Solar Aspect Angle

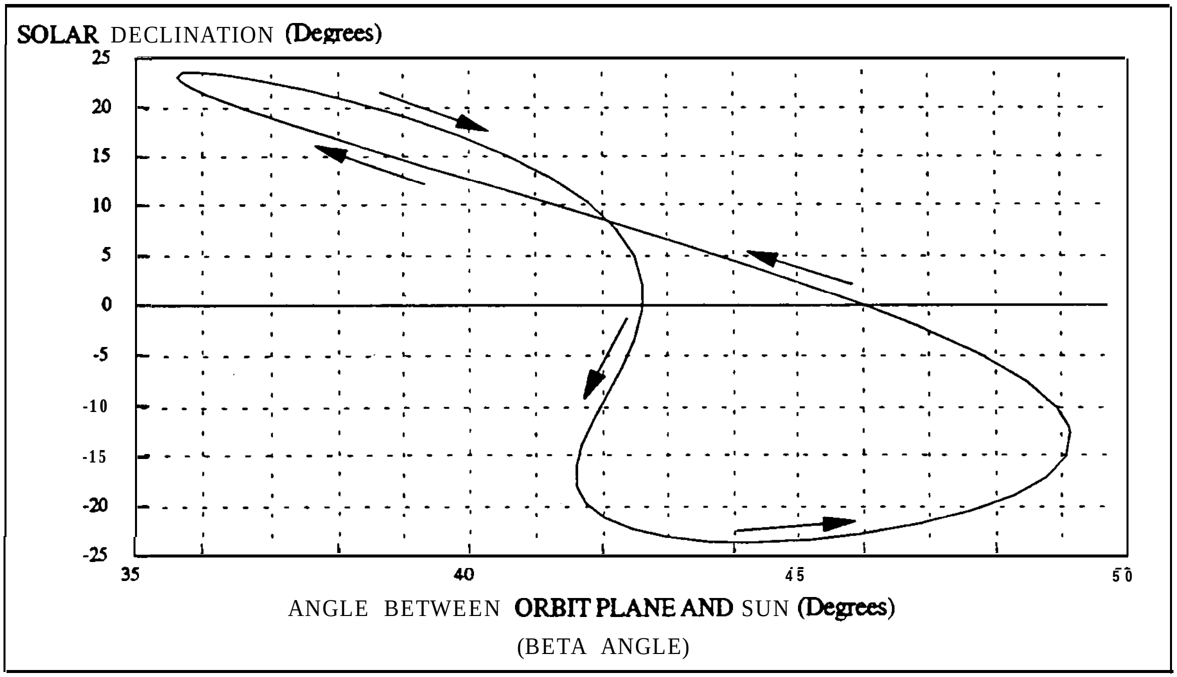

Figure 3.4/2 : Variation of Beta Angie With Solar Declination

3.4.4. External Fluxes

Launch

For METOP, the nominal instant of jettisoning the fairing will be such that the aerothermal flux is less than 500 W/m* (TBC).Solar, albedo and Earth shine heat inputs shall also be taken into account when determining the heat input into equipments during this time.

A maximum axial thermal flux of 3 kW/m² is generated for 1 second at the separation plane of the spacecraft (flux calculated on plane surface perpendicular to the spacecraft X axis) due to retrorocket firing. A maximum radial thermal flux of 0.5 kW/m² is generated during this time

Solar

The design value of the solar constant shah be 1371 ± 5 W/m² at a distance of 1.495985E+8 km (1 AU) in the absence of the Earth's atmosphere. Taking this solar constant, its uncertainty, and the distance to the Sun, Figure 3.4/3 gives the variation of solar flux during the year.

Solar Spectral Irradiance

The solar spectral it-radiance given in Table 3.4/4 shall be used for calculations which require solar irradiation data over narrow wavelength bands. The estimated error in these values is ±5% in the wavelength range 0.3 to 3.0 µm. The values are for a distance of 1 AU from the Sun and correspond to a total flux of 1371 W/m².

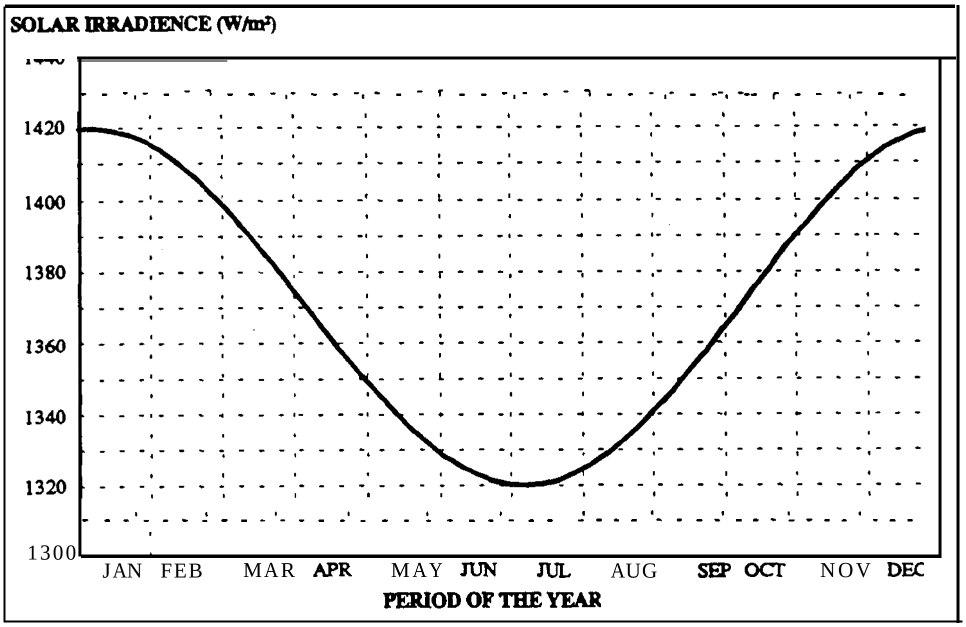

Figure 3.4/3 : Variation of Solar Flux

Earth Albedo

The global Earth albedo is the ratio of total solar radiation reflected from the Earth to the incident solar radiation. Its spectral distribution can be approximated by a black body at 5760 K.

For thermal design the Earth albedo shall be taken to vary between 03 and 0.4.

Earth shine (Earth **Radiation)**

The Earth and its atmosphere radiate in the infra-red and between 8 and 12 µm it can be approximated to that radiated by a black body at 288K. At longer wavelengths the approximated black body temperature is 218K. The emitted Earth shine is not constant over the globe as it is influenced by the temperature of the Earth's surface, (both land and sea), by the amount of cloud cover, and by the air temperature and its water content.

For thermal design the Earth shine shall be taken to vary between 216 and 258 W/m^2.

3.45. Space Sink Temperature

The radiation temperature of space is 4K.

WAVELENGTH (µm)	AVERAGE IRRADIANCE (W/m².µm)	WAVELENGTH (µm)	AVERAGE IRRADIANCE (W/m².µm)
0.22	58.3	1.10	599.9
0.24	63.8	1.20	490.4
0.26	131.7	1.30	4013
0.28	225.0	1.40	340.5
0.30	520.8	1.50	290.8
0.32	841 .0	1.60	2472
0.34	1088.3	1.70	204.7
0.36	1082.2	1.80	161.1
0.38	1134.9	1.90	127.7
0.40	1448.0	2.00	104.4
0.42	1770.2	2.10	91.2
0.44	1834.1	2.20	80.1
0.46	20935	2.30	68.9
0.48	2101.6	2.40	64.9
0.50	1967.8	2.50	54.7
0.55	1747.9	2.60	48.6
0.60	1688.2	2.70	43.6
0.65	1531.1	2.80	395
0.70	1387.2	2.90	35.5
0.75	1251.4	3.00	31.4
0.80	1121.7	3.20	22.9
0.85	1001.1	3.40	16.8
0.90	900.8	3.60	13.7
0.95	846.1	3.80	11.2
1.00	755.9	4.00	9.6

*TABLE 3.41'4: **SOLAR** SPECTRAL **IRRADIANCE** AT **1** AU FOR SOLAR CONSTANT OF 1371 **W/m²***

4. ELECTRICAL INTERFACE REQUIREMENTS

4.1. POWER SUPPLY INTERFACES

cf. Annex

42. COMMAND AND CONTROL INTERFACES

Cf. Annex

4.3. SCIENCE DATA INTERFACES

Cf. Annex

4.4. HOUSEKEEPING TELEMETRY

Cf. Annex

45. CONNECTORS AND HARNESS

Cf. Annex

5. EMC/RFC INTERFACE REQUIREMENTS

Cf. Annex

6. CLEANLINESS AND SPACE ENVIRONMENT

6.1. CLEANLINESS REQUIREMENTS AND CONTAMINATION CONTROL

Instruments shall be contamination free to the extent that they do not degrade the performance or cleanliness of the spacecraft or any other equipment, Cleanliness shall be achieved by the judicious selection of non-contaminating components and materials, use of facilities and processes that minimize contamination from these sources and use of processes to clean equipments of any accrued contaminants. A contamination control programme addressing :

- the determination of contamination sensitivity
- the determination of contamination allowance
- the determination of a contamination budget
- the development and implementation of contamination control procedures

shall be effected.

All sources of contamination that can be emitted from the instrument shall be identified and documented in the ICD.

Protective Covers

Sensitive instrument surfaces shall be covered during spacecraft integration and test, shipment, launch site processing and launch.

Provision of instrument protective cover is under the instrument responsibility. The ICD shall specify if and when protective covers (such as bags, draping materials or hard covers) are required to keep the instrument clean during AI'T phase, and the procedures for their use.

Instrument Environment

The instrument will be provided with plume flow field analyses for all thrusters : TBD.

Particulate and Molecular Cleanliness

The instruments will be integrated with the spacecraft in a class 10,000 clean room environment and maintained in that environment as much as possible during the integration and test flow.

GSE Cleanliness Requirements

Any GSE which must accompany the instrument into a clean room area must be cleaned and clean room compatible. Any GSE which must be in the vacuum chamber during thermal vacuum testing must be cleaned up and vacuum compatible.

6.2. RADIATION ENVIRONMENT

Instruments shall meet their specified performance when exposed to the radiation environment that characterizes the specified orbit for the useful lifetime (four years) and the worst case time period. These radiations are twofold :

- trapped proton and electron fluxes, high energy solar flare proton fluence,
- galactic cosmic rays and heavy ions.

63.1. Radiation Deposit Dose

The radiation dose specific to the METOP spacecraft is TBD.

The radiation tolerances for each instrument component design (integrated dose) shall be included in the specific instrument ICD.

62.2. Single Event Upset (SEU) and Latch-Up Effects

Those are instantaneous effects produced by energetic particles.

Instruments shall be capable of withstanding and recovering from SEUs and transients induced by the singular or combined effects for cosmic rays, solar flares and geomagnetically trapped protons.

The fluxes specific to the METOP spacecraft are TBD.

The fluxes considered to assess the instrument component criticality with regard to SEU and latch up shall be documented in each specific ICD.

63. SPACE ENVIRONMENT CONSTRAINTS

63.1. Meteoroid and Space Debris

METOP in a low Earth orbit is exposed to a certain flux of micrometeoroids and man-made space debris. Collisions with these particles can take place at relative velocities of tens of km/s. All flight assemblies shall be designed to function within specification during an after exposure to the meteoroid and space debris fluence for the four year operation time.

The meteoroid and space debris fluence specific to the METOP spacecraft is TBD.

The assumptions considered for each instrument with respect to these fluxes shall be included in each specific instrument ICD.

63.2. Atomic Oxygen

The spacecraft motion through the residual atmospheric atomic oxygen generates a flux to the spacecraft surfaces. The incident atomic oxygen can lead to chemical reactions, and hence changes of the surface characteristics, in terms of thermo-optical, structural, outgassing or electrical properties.

All surface materials exposed to atomic oxygen shall be designed to function within specification during exposure to the atomic oxygen fluence.

The way this effect is addressed for the instrument design shall be documented in each specific ICD.

7. INSTRUMENT DESIGN AND VERIFICATION REQUIREMENTS

This section establishes the verification requirements for the qualification and flight certification of the instrument units giving specific test levels and durations and describing acceptance test and analytical methods for implementing the requirements.

The Instrument Supplier shall prepare a Verification Plan which defines the tests and analyses and that collectively demonstrates that hardware and software complies with the requirements.

The Verification Plan shah show the overall approach to accomplish the instrument qualification and acceptance pmgramme. When appropriate the interaction of the test and analysis activity shall be described.

All items to be flown on the spacecraft have to be qualified. The qualification demonstrates that the product meets the design and performance requirements. An item that has been already qualified needs only acceptance testing when qualified to levels same as or greater than specified for METOP. The acceptance demonstrates that the particular item is functional in its nominal environment.

7.0. DEFINITIONS

Design Qualification Verification

Tests and analyses intended to demonstrate that the item will function within performance specifications under simulated conditions more severe than those expected from ground handling, launch and orbital operations. The purpose is to uncover deficiencies in design and method of manufacture and is not intended to exceed design safety margins or to introduce unrealistic modes of failure.

Acceptance Verification

Tests intended to demonstrate that hardware is acceptable for flight. It also serves as a quality control ***screen*** to detect deficiencies and normally to provide the basis for delivery of an ***item*** under terms of a contract or agreement.

Functional Tests

The operation of a unit in accordance with defined operational procedures to determine that performance is within the specified requirements.

Performance **Verification**

Determination by test analysis or a combination of the two that the complete instrument or instrument unit can operate as intended in a particular mission : this includes proof that the &sign of the complete instrument or instrument unit has been qualified and that the particular item has been accepted as compliant to the &sign and ready for flight operations.

Thermal Balance **Test**

A test conducted to verify the adequacy of the Thermal Model the adequacy of the thermal design and the capability of the thermal control system to maintain thermal conditions within established mission limits.

Thermal Vacuum Test

A test to demonstrate the validity of the design in meeting functional goals. It also demonstrates the capability of the test item to operate satisfactorily in vacuum at temperatures based on those expected for the mission. The test can also uncover latent defects in design, parts and workmanship.

Static Loads

The maximum combination (longitudinal and lateral) of static loads which acts on an instrument during the various segments of the flight profile. It consists of steady state accelerations (e.g. due to engine constant thrust or lateral wind loads) and quasi-static loads which are structure borne loads generated by the launch vehicle in the low frequency (less than 100 Hz) range (e.g. engine cut-off loads or wind gusts).

Sinus Vibration Test

A test to demonstrate that the instrument can withstand the mechanical environment of the low frequency (less than 100 Hz) sinusoidal and transient vibrations.

Acoustics / Random Vibration

An environment induced by high-intensity acoustic noise associated with various segments of the flight profile : it manifests itself throughout the instrument in the form of directly transmitted acoustic excitation and as structure-borne random vibration excitation.

Electromagnetic Compatibility (EMC)

The condition that prevails when various electronic devices are performing their functions according to design in a common electromagnetic environment.

Electromagnetic Interference (EMI)

Electromagnetic energy which interrupts, obstructs, or otherwise degrades or limits the effective performance of electrical equipment.

Electromagnetic Susceptibility

Undesired response by a component instrument or system to conducted or radiated electromagnetic emissions.

7.1. TESTING

7.1.1. Test Procedures

For each test defined in the verification plan (e.g. EMC, vibration, electrical, thermal...), the instrument supplier shall provide a detailed procedure. A final report for each test shall also be provided.

7.1.2. Test Matrix

In addition to the Verification Plan, the instrument supplier shall prepare a Test Matrix that summarizes all the tests that will be performed on each instrument unit. The purpose of the matrix is to provide a ready reference to the contents of the test programme in order to prevent the deletion of a portion thereof without an alternate means of accomplishing the objectives. It has the additional purpose of ensuring that all flight hardware has seen environmental exposures that are sufficient to demonstrate acceptable workmanship. In addition, the matrix shall provide traceability of the qualification heritage of hardware. All flight hardware, spares (and prototypes when appropriate) shall be included in the matrix.

7.1.3. Test Sequences

No specific environmental test sequence is required, but the test programme should be arranged in a way to best disclose problems and failures associated with the characteristics of the hardware and the mission objectives. It is strongly recommended that the vibro-acoustic test precede the thermal vacuum test unless there is an overriding reason to reverse that sequence.

7.1.4. Test Condition Environment

Unless otherwise specified, all system, module or equipment fabrication, handling and tests shall be made at ambient conditions as specified below. These environmental conditions shall be controlled and recorded.

- Temperature TBD
- Humidity TBD
- Pressure TBD
- Cleanliness TBD

For transportation and storage, the following environment applies :

- Temperature TBD
- Humidity TBD
- Pressure TBD

7.1.5. Test Condition Level Tolerances

The test tolerances, unless otherwise specified, are :

- Temperature TBD
- Humidity TBD
- Pressure TBD
- Sinusoidal vibration TBD
- Random vibration TBD
- Static force TBD
- Acoustic TBD
- Electromagnetic Compatibility TBD
- Magnetic Properties TBD
- Mass properties (weight, centre of gravity, moments of inertia) TBD
- Mechanical shock TBD

7.1.6. Test Equipment Accuracy and Calibration

The accuracy of the instrument and test equipment used to control or monitor the test parameters shall be verified periodically by calibration procedures.

7.2. TEST REQUIREMENTS

7.2.1. Electrical Functional Test Requirements

7.2.1.1. Electrical Interface Tests

Before the integration of an assembly, electrical interface tests shall be performed to verify that all interface signals are within the specified limits.

7.2.1.2. Full Performance Test

The Full Performance Test (FPT) shall be a detailed demonstration that the hardware and software meet their performance requirements within allowed tolerances. The test shall demonstrate operations of all redundant circuitry. It shall also demonstrate satisfactory performance in all operational modes. The initial FPT shall serve as a baseline against which the results of all later FPTs can be readily compared .

The test shall also demonstrate that, when provided with appropriate stimuli, performance is satisfactory and outputs are within allowed limits.

7.2.1.3. Limited Performance Test

The Limited Performance Tests (LPT) shall be a subset of the FPT and shall be performed before, during, and after environmental tests, as appropriate, in order to demonstrate that functional capability has not been degraded by the environmental tests. The limited tests may also be used in cases when comprehensive performance testing is unwarranted or impracticable. Specific items on which it is intended that Limited Performance Tests will be performed, shall be listed in the Verification Plan. The Limited Performance Tests shall demonstrate that the performance of selected hardware and software functions is within allowed limits.

7.2.1.4. Limited Life Time Elements

A life test programme shall be performed on electrical or electromechanical elements that have limited lifetimes. The Verification Plan shall address the life test programme, identifying the electrical or electromechanical elements that require such testing, describing the test hardware that will be used and the test methods that will be employed.

7.2.2. EMC Test Requirements

The requirements laid down § 5 shall be verified by tests, following as far as possible the standard methods (TBD).

7.2.2.1. Qualification Test

For the qualification of the experiment the "Qualification" Model shall be subjected to the full EMC test sequence outlined below :

- ESD Test,
- Bounding, grounding, isolation measurement
- Conducted emission
- Conducted susceptibility
- Radiated emission
- Radiated susceptibility

7.2.2.2. Acceptance Test

This test shall be accomplished on all deliverable hardware. Acceptance level testing shall comprise the verification of :

- Bounding
- Isolation
- Grounding and conductivity test of space expose surfaces
- Conducted interference tests
- Structure induced noise test.

7.2.3. Mechanical and Structural Test Requirements

A series of tests and analyses shall be conducted to qualify the design of the hardware, to demonstrate specified factors of safety, interface compatibility, workmanship and compliance with associated launch authority safety requirements.

7.2.3.1. Structural Testing : Quasi-static Test

The primary objective of this test is to demonstrate that the load carrying structure withstands the flight limit loads without rupture, collapse, damage or permanent deformation.

The test is not required in the following cases :

- if analysis demonstrates positive margin of safety against the design limit loads accounting for the safety factors of 1.25 and 2 at respective yield and ultimate for metallic structures, or a safety factor of 3 for non-metallic structures
- for hard mounted units, the test being covered by the vibration tests described in § 7.2.3.3.

Parts and subassemblies which do not fall in these cases shall be proof tested at 1.25 times the limit loads.

7.2.3.2. Structural Testing : Dynamic Model Validation

The structural mathematical model (Cf. § 2.5.4.) shall be verified by test and updated accordingly.

All the modes observed during the test will be identified by their mode shape, their frequency and their damping ratio. The predicted modes of the model shall be correlated / updated against modes whose effective mass is greater than 5% of the rigid body mass. After validation, the model shall still describe the modes up to 150 Hz.

This validation can be done by a specific modal survey test or a low level sine test whichever is the most practical.

7.2.3.3. Structural Testing : Vibration Tests

Sinusoidal and Random vibration tests are required to demonstrate that the experiment units can survive mechanical stresses arising during launch as well as the spacecraft AIV programme.

Facility

Test facility Cleanliness

Fixture Requirements

Configuration

The instrument shall be vibrated in the launch configuration. Test adaptors and/or non-flight items shall be removed prior to vibration. For the deployed configuration, TBD (Cf. § 2.5.4.).

Vibration and Control Instrumentation

Recording Instrumentation

Qualification Vibration Testing

Items to be flown on METOP shall be mechanically qualified at instrument level with the following tests as a minimum :

- High Level Sine, but note that Sine Burst is acceptable instead of High Level Sine for items whose first natural frequency is above 100 Hz.
- Random Vibrations

Levels are defined below :

a) High Level Sine Sweep

This Sine test shall be conducted in all three axes with the following levels depending on the natural frequency of the equipment :

- Equipment with first natural frequency above 100 Hz

5 to 18 Hz	±11 mm
18 to 60 Hz	±15 g
60 to 100 Hz	±6 g

Sweep rate: 2 octaves/minute

(Note that these values are TBC, different values may be specified after the ESA launch load restitution analysis to be performed in September 1994)

- Equipment with first natural frequency below 100 Hz

  Same levels and durations as above but with narrow band notching. The narrow band notching will only be allowed in the resonances of the equipment. No wide band notching is allowed. In particular, wide band reduction up to 100 Hz will not be allowed. In all cases, the response at centre of mass of the equipment shall be limited to the acceleration level definition to mass of the item as per table of Design Loads given in § 2.5. The notching approach must be detailed and the profile approved before the test.

b) Sine Burst

Sine Burst tests in all three axes of one second duration per axis an to a level of 15 g may be performed instead of High Level Sine for equipment whose first natural frequency is above 100 Hz. The test frequency shall be less than 20 Hz (TBC).

c) Random Vibration

The random vibration test shall be conducted in all three axes with a duration of 2.5 minutes per axis.

- Perpendicular to mounting plane:

20 to 100 Hz	+3 dB/Octave
100 to 400 Hz	$PSD = 0.05 \frac{M+20}{M+1}$ g²/Hz, or 0.07 g²/Hz whichever is higher
400 to 2000 Hz	-3 dB/Octave

- Two axes parallel to mounting plane:

20 to 100 Hz	+3 dB/Octave
100 to 200 Hz	$PSD = 0.05 \frac{M+20}{M+1}$ g^2/Hz, or 0.07 g^2/Hz whichever is higher
200 to 2000 Hz	-4 dB/Octave

Acceptance Vibration Testing

For equipments to be flown on METOP the only mechanical test that is compulsory shall be random vibration. Random vibration acceptance levels are equal to the qualification ones but divided by 1.25 for accelerations and $1.25^2 = 1.56$ for g^2/Hz. The duration shall be 2 minutes per axis.

- Perpendicular to mounting plane:

20 to 100 Hz	+3 dB/Octave
100 to 400 Hz	$PSD = 0.032 \frac{M+20}{M+1}$ g^2/Hz, or 0.045 g^2/Hz whichever is higher
400 to 2000 Hz	-3 dB/Octave

- Two axes parallel to mounting plane:

20 to 100 Hz	+3 dB/Octave
100 to 200 Hz	$PSD = 0.032 \frac{M+20}{M+1}$ g^2/Hz, or 0.045 g^2/Hz whichever is higher
200 to 2000 Hz	-4 dB/Octave

Instruments qualified in the past to METOP equivalent or higher values will need only acceptance testing.

To be considered as being already qualified, an Instrument Contractor shall provide documentary evidence of the test and analysis that were performed during the instruments original qualification.

Shock

The performance of specific tests to simulate, at instrument level, the behaviour of the instruments against externally generated shocks is not compulsory.

The instrument shall be designed to withstand the loads generated by the separation of the launcher. These shocks are TBD (Cf. ARIANE 4 & 5 Users Manual).

The instrument shall also be designed to withstand the shock loads generated by the deployment of other items on-board the spacecraft. This shall be demonstrated by analysis at instrument level and by testing at spacecraft or module level.

The instrument shall be able to withstand its own shocks. This shall be verified by test at instrument level during functional test of the instrument.

7.2.3.4. Acoustic Test

The acoustic noise test shall be performed in addition to the vibration test on those units having a large surface and high frequency sensitivity where the acceleration responses are mainly due to direct acoustic inputs rather than to structure-borne inputs. The units concerned shall be agreed between the instrument supplier and the METOP project team.

Test Set-Up

Test Description

Test Levels

Test Sequence

TBD (Cf. ARIANE 4 & 5 Users Manual)

7.2.3.5. Mechanisms Verification

Tests shall be performed to check mechanism performances in both launch and operational configurations. Mechanisms can be considered as structures as far as strength and stiffness tests are concerned, and their design shall be verified along similar lines applicable to other structural components. As a reference, the following test sequences are applicable :

- functional tests (before and after thermal vacuum exposure)
- mechanical environment tests
- life test.

7.2.4. Thermal Test Requirements

An appropriate set of tests and analyses shall be conducted to demonstrate the following instrument capabilities :

- the satisfactory performances within vacuum and mission established thermal limits
- the thermal design compatibility in order to maintain hardware interfaces within specified limits
- the quality of workmanship and materials of the hardware to pass thermal cycle test.

7.2.4.1. Thermal Cycle Vacuum Tests

The thermal cycle vacuum tests are required to evaluate and demonstrate the functional performance of each unit under the extreme and nominal modes of operation while in simulated vacuum and at temperatures more extreme than predicted for the orbit conditions. The purpose of the more severe temperature stress is to demonstrate a design safety margin and to accelerate failure in marginal design.

Cycling between temperature extremes has the purpose of checking performance at other than stabilized conditions and of causing temperature gradient shifts, thus inducing stresses intended to uncover incipient problems. A minimum of four (4) cycles shall be performed during both qualification and acceptance tests.

Facility TBD

Test Method : TBD

Test Level and Duration : TBD

Functional Test : TBD

TV Test Sequence : TBD

7.2.4.2. Thermal Balance Test

The thermal balance test simulates extreme and nominal conditions to verify the instrument thermal control system. The number of energy balance conditions simulated during the test shall be sufficient to verify the thermal design. The exposure shall be long enough for the test item to reach stabilization so that the temperature distribution in steady-state condition may be verified.

Test Method : TBD

Test Level and Duration : TBD

8. GROUND SUPPORT EQUIPMENT

8.1. MECHANICAL GROUND SUPPORT EQUIPMENT

8.1.1. Introduction

All mechanical ground support equipment needed to support the instrument during system level AIV operations, shall be provided by the instrument supplier, together with written procedures as to its used. The above mentioned activities shall be supported with trained personnel who are familiar with the instrument and the MGSE equipment.

MGSE shall be supplied in sufficient quantities (one set per instrument model) to ensure efficient integration and test on the various spacecraft.

The type of equipment normally classed as MGSE incudes such dedicated items as :

- instrument handling jigs and/or support frames
- special tools
- alignment devices
- containers or enclosures used to protect the instrument against environmental extremes and contamination during transportation
- test fixtures, adapters, stands and lifting equipment
- purging equipment

8.1.2. Test Fixtures and Configuration

TBD

8.1.3. MGSE Cleanliness

TBD

8.1.4. Transport Containers

A dedicated container shall be supplied for each instrument model and ground support equipment. These containers shall provide adequate protection against the environment, both in storage and in transport;

For sensitive items (e.g. focal camera units...) shock, humidity and temperature recorders shall be installed during transport.

8.1.5. Load Factors for Handling and Transportation

TBD

8.2. ELECTRICAL GROUND SUPPORT EQUIPMENT

CF. Annex.

9. GROUND OPERATIONS

The system level ground operation phase covers all spacecraft ground operations defined in the METOP AIT/AIV programme, together with the launch campaign.

The sequence of ground operations is as follows :

- delivery of instrument and associated support equipment to the PLM integration site followed by an incoming inspection and abbreviated bench level functional test
- interface and integration verification
- spacecraft system level electrical performance validation and environmental tests including flight control centre compatibility tests (and instrument specific tests if required).
- launch site operations including spacecraft final electrical checks and integration onto the launch vehicle.

9.1. SYSTEM VERIFICATION PROGRAMME - MODEL PHILOSOPHY

The METOP spacecraft verification programme consists in validating the spacecraft system performances by a set of tests and analytical methods. This concept achieves a full qualification of the spacecraft system design based on a programme of two development models (SM/EM) and flight model spacecraft, described herewith.

Only the PFM proto-flight model (METOP-1) and FM flight model (METOP-2) will be built at flight standard using hi-rel parts.

To minimize the risks of such approach the flight models are preceded by SM and EM development models, representative of flight standard but not equipped with hi-rel parts. Both models participate to the qualification process.

The Structural Model (SM) is used to perform first a separate design validation of the payload module structure (service module TBC), followed by a mechanical qualification at satellite level.

The Engineering Model (EM) is dedicated to the electrical verifications at system level and to the validation of all electrical test equipment and test methods including full EMC testing procedures. This will be performed at payload module level (and at spacecraft level if necessary but on the PFM model).

The Proto-Flight Model (PFM), fully flight representative, is submitted to all functional, performance, electrical and environmental tests necessary to finally demonstrate the qualification of the design, its ability to fulfil the mission requirements and to comply with the workmanship quality requirement.

The Flight Model (FM) is also fully flight representative and is submitted to the same test as PFM but at an acceptance level.

To perform such a validation process at system level, the following instrument models shall be supplied.

9.1.1. Instrument Structural Model (SM)

An instrument structural model shall be supplied as soon as the instrument first frequency is below 100 Hz. On the contrary, the instrument SM will be replaced by dummies that will be supplied by the spacecraft contractor (TBC).

The instrument structural model built-standard shall be representative of the flight-model in the following domains :

- physical parameters
  . mass, centre of mass, moment of inertia, dimension
- interface parameters
  . mounting interfaces shall be similar to the flight standard as far as flatness, fixation hole position, diameter and tolerance are concerned
  . materials shall be the same as used for flight hardware
  . instrument dummy harness
  . alignment mirror
  . connector
  . purging inlet
- mechanical parameters
  . the unit shall undergo qualification testing sufficient to demonstrate that the design performances meet the strength and stiffness requirements

9.1.2. Instrument Engineering Model (EM)

The instrument engineering model is an electrical model used to validate the electrical design (functional and EMC), the GSE design and procedures. This validation is achieved through functional/performance and EMC.

To cope with this approach, the engineering model shall be representative of the flight model in the following domains :

- mechanical parameters

  Mass, dimension, mounting interface, connector type and location shall be representative of flight standard
- thermal parameters

  The thermal control should be fully representative with radiators, radiator support, heat pipes, external surface for EMC/ESD testing, since the payload module EM will under go thermal tests (TB/TV).

- electrical parameters

  All hardware and inter-instrument harness shall be electrically and functionally representative of the flight standard (save use of high-reliability parts). EMC and functional tests shall be demonstrated by the successful completion of electrical tests (cf. § 7).

- connectors

- EGSE

  The EGSE hardware shall be similar to the one used to support the flight model system tests and shall contain a similar control test files/command data base.

Note that the use of a Proto-Flight Model or Flight Model instead of Engineering Model can be considered.

9.1.3. Instrument Proto-Flight Model (PFM)

The instrument flight model shall be of a standard compliant with all the requirements of the present document (GICD) and the description of its corresponding ICD. It shall have successfully undergone a full programme of acceptance level testing and verification prior to delivery.

The model shall be fully calibrated and have passed thermal vacuum test prior to delivery.

9.1.4. Instrument Flight Model (FM)

The Instrument Flight Model (FM), which is a recurring version of the Instrument Proto-Flight Model (PFM), shall be such that no extra qualification test is needed following the PFM qualification campaign. Its performances and characteristics shall then remain inside the domain defined for the first model.

9.1.5. Flight Spare Model

TBD (Cf. § 9.1.3.).

9.2. DELIVERY OF THE INSTRUMENT TO THE AIV SITE

TBD

9.3. INSTRUMENT INTEGRATION

TBD

9.4. PURGING REQUIREMENTS

The platform purging system is TBD.

Instrument purge requirements, including type of purge gas, flow rate, gas purity specifications, filter pore size, type of desiccant (if any), and whether interruptions in the purge are tolerable shall be documented in the ICD.

The following information on the purge connection shall be provided :
- type and position of purge valve and cap ;
- type and position of vent or bleed valve.

9.5. GROUND ENVIRONMENTAL CONDITIONS

Integration and Nominal Test Environment

Special Environmental Requirements

Unprotected Environment

9.6. LAUNCH OPERATIONS

TBD

10. FLIGHT OPERATIONS

10.1. OVERVIEW

The METOP mission will consist of a three-axis stabilised spacecraft placed into a Sun synchronous orbit around the Earth. It will maintain the nominal fine Earth pointing throughout its operational lifetime except during those periods when orbit control manoeuvres are in progress (TBC). The spacecraft operational design lifetime is four years with on-board consumables for extension to six years.

10.2. ORBITAL PARAMETERS

10.2.1. Operational Orbit

METOP will be placed into the following reference orbit :

- Type : polar Sun-synchronous
- Repeat Cycle : 5 days (14 + 1/5)
- Local Solar Time : 09:00 A.M. descending node

The mean Kepler elements are :

- Semi Major Axis : 7197.939 km
- Eccentricity : 0.001165
- Inclination : 98.704 deg
- Right Ascension of the Ascending Node : 55.835 deg
- Argument of Perigee : 90.000 deg

This reference orbit will be maintained within the following tolerances :

- mean local solar time : ± 5 minutes
- orbit track repeatability : ± 5 km at any latitude.

10.2.2. Pointing Characteristics

Nominal Mode

The commanded absolute pointing will nominally be :

- (-Zs) pointing towards nadir
- (-Ys) pointing towards the velocity

The platform will provide either a fine pointing mode (FPM) or a yaw steering mode (YSM). In such modes, the pointing performances of the instrument interface reference frame are the following :

Absolute Pointing Error	0.12/0.14/0.17 deg (3σ, 0-4 Hz, FPM, TBC) 0.14/0.18/0.26 deg (3σ, 0-4 Hz, YSM, TBC)
Absolute Measurement Error	0.14/0.17/0.25 deg (3σ, 0-4 Hz, YSM, TBC)
Absolute Rate Error	0.005 deg/s (3σ, 0-4 Hz, TBC)

Note that the above performances are not guaranteed during orbit control manoeuvres (altitude maintenance). The characteristics are then :

Frequency of manoeuvres	TBD
Duration of manoeuvre	TBD
Pointing Performances	TBD

In addition to the nominal FPM or YSM pointing performances, co-alignment between instrument interface feet will be realised (Cf. § 2.4.3.).

Sun Pointing

As an ultimate safety level, a specific safe mode is implemented. Its characteristics are :

- (+ Zs) pointing towards the Sun
- Spin around this axis at 0,6 deg/s (TBC)

In such a mode, the pointing performances (with respect to the Sun direction) are :

- ± 15 deg during Sun-lit phases,
- ± 30 deg during shadowing phases
- delay between shadowing and Sun-lit performances : 300 s

10.3. MISSION OPERATION PHASES AND MODES

10.3.1. Mission Phases

During its lifetime the satellite will be operated according to mission phases which can be broken down into further sub-phases :

- **Integration and Test Phase**

  The integration and test phase will encompass all satellite processing activities from equipment assembly up to and including satellite acceptance testing at the Contractor site. It will end at the Flight Acceptance Review.

- **Transportation/Launch Site Checkout Phase**

  This phase will start subsequently to satellite acceptance at the Contractor site and end with final countdown on the launch site. The proper launch phase will begin at the instant of switching the power subsystem to on-board batteries and will end at satellite/launch vehicle separation.

- **Acquisition Phase**

  This phase will start at the end of the launch phase and end once the satellite has acquired its operational attitude and orbit with its appendages deployed. An initial acquisition sequence leading to a 'safe state', is followed by a final acquisition period.

- **Commissioning Phase**

  This phase will start once the nominal attitude and orbit have been acquired and will cover the time that subsystems and instruments are checked out, and will end when the payload is operational.

- **Routine Phase**

  This phase will start at the end of the commissioning phase and will cover the time that the instruments are operational and the times when orbit maintenance manoeuvres are performed.

Typical phase durations are given in Table 10.3/1.

PHASE	DURATION
Transportation/Launch Site Check-out	4 months
Launch Phase	30 minutes
Acquisition Phase	Several Days
Commissioning Phase	Up to 3 months
Routine Phase	4 years including the commissioning phase

TABLE 10.3/1 : TYPICAL MISSION PHASE DURATIONS

10.3.2. Platform Modes

This section describes the Platform operational modes. A summary table is presented at the end of this section.

10.3.2.1. Inert Mode

This mode is used for handling, transportation and storage of the Platform. It is the first mode on the launch pad. In inert mode, the following apply :

- All equipments are off,
- No electrical current is circulating in the platform,
- No RAM software is loaded,
- All instruments are off.

10.3.2.2. Ground Mode

This is an intermediate mode between the inert mode and the launch mode in which the platform is switched on and initialised for launch. In this mode :

- Platform power is supplied via umbilical from ground,
- The PLM remains off with the exception of survival power and the payload controller which is switched on but is in standby mode with its software not active. This is the PLM launch and early orbit mode (LEOM).

10.3.2.3. Launch Mode

The launch mode is initiated by ground telecommand at the beginning of the launch phase (typically several minutes before lift off) when the power supply is switched to the SVM batteries. The platform remains in this mode throughout the launch itself until shortly after separation from the launch vehicle. In launch mode the platform performs the following functions :

- Platform power supply via batteries,
- Maintenance of the PLM in LEOM mode.

All instruments shall be off during LEOP.

10.3.2.4. Attitude Acquisition Modes

Following launch vehicle separation, the acquisition sequence (the pyro sequence) is automatically initiated by launch vehicle separation signals. During this sequence the gyros are switched on, the array is released, array primarily deployment occurs and then the platform passes automatically through four AOCS acquisition modes : RRM, CAM, FAM1 and FAM2 in which the angular rates imparted at launch vehicle separation are reduced and the nominal operational attitude is acquired. Following attitude acquisition the activation sequence continues automatically with solar array secondary deployment followed by SADM release and initiation of array rotation. As a backup, the pyro sequence may be started and performed by ground command. The PLM remains in LEOM mode throughout this mode.

Note that the satellite autonomy in LEOP shall be at least of three full orbit periods.

10.3.2.5. Initial Transition to Pre-Operational Mode

After solar array secondary deployment and initialisation of rotation, the remaining SVM subsystems are activated and the payload manager automatically initialized. The platform then waits in FAM2 mode for further ground commands.

The PLM can then be activated and the operational appendages deployed by ground command. After PLM activation, the PLM is in operational standby mode in which :

- the payload manager is on with its S/W activated monitoring and controlling the PLM,
- the PLM OBDH bus is active and the Remote Terminal Units (RTU) are on,
- the PLM thermal control is S/W controlled,
- the Payload Power Distribution Units (PPDU) are operational,
- the instrument hardware units are off although some Instruments Control Units (ICU) may be on,
- all major payload appendages are deployed,
- the PLM communications system and Data Management System (DMS) are off or in standby mode (with the exception of the PMC, OBDH and RTU's).

10.2.3.6. Operational Mode

During operational mode, the platform will be in one of the following two pointing modes :

- Fine Pointing Mode (FPM), in which the platform maintains three axis local normal pointing;
- Yaw Steering mode (YSM), in which the platform maintains three axis local normal pointing as for FPM but with an additional yaw steering bias of the reference frame to compensate for the apparent drift of the sub-satellite point due to the Earth's rotation.

The fine pointing mode is the intermediate attitude pointing mode for the satellite between fine acquisition and yaw steering mode. The yaw steering mode is the routine attitude pointing mode for the satellite.

In these operational modes the platform performs all the functions required for routine operations, including payload monitoring and control. The Payload Module is in operational mode with :

- the payload manager on with its S/W monitoring and controlling the PLM,
- all PLM subsystems on and operating nominally,
- thermal control S/W controlled,
- instruments on and operating.

The transition from FAM2 to the operational modes is initiated by ground command.

10.2.3.7. Orbit Control Mode (OCM), Fine Orbit Control Mode (FCM)

These are AOCS modes active during orbit maintenance manoeuvres. Normal mission operations may continue if the mission requirements are compatible with the loss of pointing accuracy and stability during these manoeuvres. This is excluded for OCM, and may be compatible for FCM (TBC after further analysis).

In FCM, the platform performs adjustments of orbital parameters by thrusts in the orbit plane of limited durations (less than 1 minute). The transitions from Operational Mode to OCM or FCM are initiated by ground commands while transitions from OCM or FCM back to Operational Mode are performed automatically or by time tagged command.

10.2.3.8. Back Up Modes

Several back up modes exist for the platform :

- Payload Fall Back Modes

  In the event of detection of a PLM failure, the platform shall autonomously switch the PLM into one of the following modes depending on the criticality of the failure :

  - Operational standby mode or Payload safe mode.

    In the payload safe mode :

    . the payload manager is in hold (suspend) mode,

    . the PLM DMS and communications systems are off,

. the PLM thermal control is hardware controlled (by thermostats),

. the PLM power distribution system is powered and nominal power is available from the SVM,

. the instruments are in a safe condition (ICU's off, hardware off).

Return to PLM operational mode from these back up modes is under ground control.

- SVM Back Up Earth Pointing Modes (TBC)

  The SVM provides other back up modes which are characterized by :

  - the attitude remains Earth pointing
  - power is provided to the PLM but possibly at a lower level than for nominal operations,
  - the platform and its payloads are maintained in a safe state.

- Safe Mode

As an ultimate safety level, the safe mode ensures the satellite survival and autonomy, waiting for a reconfiguration from ground. In this mode, the platform performs the minimal functions for platform survival and the PLM is in Payload Safe Mode. Thermal control during safe mode is performed by thermostatically controlled heaters which are different to those used during nominal operations.

Phases and modes overview

Mission Phase	Platform Mode	Service Module Mode	Payload Module Mode	Definition
Transportation storage	Inert mode	Inert mode	Inert mode	Service Module and Payload Module unpowered
Launch site	Ground mode	Ground mode	Launch and Early Orbit Mode (LEOM)	Service Module powered by Ground SVM S/W in TM/TC mode Payload Module OFF except for PLM computer and equipment necessary for survival (heaters etc.) PLM S/W not active
Launch	Launch mode	Launch mode	LEOM	Service Module autonomous Platform powered by batteries
Acquisition	Rate Reduction Mode (RRM)	Rate Reduction mode (RRM)	LEOM	Reduce angular rates to 0.3°/s
	Coarse Acquisition Mode (CAM)	Coarse Acquisition mode (CAM)		Acquire Earth pointing (Pitch, Roll)
	Fine Acquisition Mode 1 (FAM1)	Fine Acquisition mode 1 (FAM1)		Maintain the geocentric pointing
	Transition to pre-operational mode			
	Pre-operational (FAM2)	FAM2	Deployment	PLM S/W activated Appendages deployed Subsystems activated
	Operational	FPM	Operational/stand-by mode	Initiation of fine pointing/yaw steering
	Orbit Control mode (OCM)	OCM	Operational/stand-by mode	Perform manoeuvres to acquire reference orbit
Commissioning phase	Operational	FPM or YSM	Operational	Satellite functional verification
	Fine orbit Control Mode	FCM	Operational	Orbit maintenance manoeuvres
Routine	Operational	FPM or YSM	Operational	Operational Mode
	Orbit Control Mode (OCM) fine orbit control mode (FCM)	OCM FCM	Operational/standby	Perform manoeuvres for orbit corrections
	Payload fall back modes	FPM or YSM	Operational/standby	PLM on and activated but operations suspended until ground intervention
			Payload Safe Mode	- S/W suspended - PLM DMS and comms systems off - Instruments off (ICUs and H/W)
	RRM, CAM, FAM1, FAM2 (Reacquisition)	RRM, CAM, FAM1, FAM2	Operational/Standby	- Service Module in Earth reacquisition
	Safe Mode	Safe Mode	Payload Safe Mode	- Sun Pointing - TM/TC by PROM Software
	Return from safe mode	Return from safe mode	Payload Safe Mode	- Sun pointing - TM/TC by nominal RAM Software

10.4. OPERATION CONSTRAINTS AND RESPONSIBILITIES

All instruments will be continuously operated over the orbit. However payload operation with only a sub-set of instruments working is possible. Special attention shall be paid during the instrument design to the commandability and observability aspects.

10.4.1. Commandability

All the telecommands shall be compliant with the general rules defined in the Telecommand Plan (TBD).

The number of telecommands necessary to perform any instrument operation shall be minimized.

Requirements on software : TBD

10.4.2. Observability

The telemetry shall report the occurrence of the failure/fault/error and all mode and configuration changes executed in response to the failure event. Detection and reporting of data corruption, including the ones due to SEU, shall be performed.

The instruments shall give the platform the opportunity to detect and isolate the following failures :

- electrical overload on the power supply lines, due to hard or partial short circuits or otherwise non nominal power consumption in the instrument, including the pyro lines ;
- corruption of protocol, formats, bit pattern... on data interfaces ;
- non execution of the satellite emitted commands ;
- signal outage on instrument generated data lines ;
- hard or partial short circuit on data / command lines

When redundancy is foreseen, the telemetry shall enable the ground to identify the failed item, and to identify and utilise the healthy remainders for reconstruction of a back-up mode. Any reconfiguration shall end with a fully defined configuration status of all involved units and software.

10.4.3. Information Provided by the Platform

The following information will be made available from the spacecraft to the instruments :

- Orbit state vector

  contents : osculating Keplerian elements
  frequency : 1 Hz
  Time delivery : not later than one second before its validity.
  The accuracy will be 600 m (3σ) along track (assuming perfect initial conditions and perfect solar activity prediction).

- Time-tagged equator crossing and day/night flags, delivered not earlier than 2 sec before its validity, and not later than 1 sec after its validity.

- Depointing signal word, whenever the platform mispoints of ± 1 deg, so that the instruments can achieve a safe state.

10.5. INSTRUMENT OPERATION MANUAL

This document shall contain all the information needed for the operation of the instrument, in particular :

- the instrument configuration and modes during the various mission phases
- the operational scenario considered
- the ground segment support requested
- the instrument data base provided.

It shall be prepared according to the standard layout described in Appendix I.

Special attention shall be paid to the instrument Thermal Control Subsystem operations.

11. PRODUCT ASSURANCE AND RELIABILITY

Product Assurance Plan

Quality Assurance

Reliability Assurance

Safety Assurance

Component Quality, Selection and Procurement

Material and Process Selection and Control

Acceptance of Deliverable Items

12. PROGRAMME AND SCHEDULE

Overall Programme

Project Reviews

Instrument Reviews

Schedule

Deliverable Items

Deliverable Documentation.

APPENDIX I

STRUCTURE AND CONTENTS OF

THE INSTRUMENT OPERATIONS MANUAL

1. GENERAL DESCRIPTION AND SCIENTIFIC OBJECTIVES

1.1. GENERAL

Section 1 should provide a general description of the instrument, its scientific objectives, and the data collected during the mission.

1.2. DESCRIPTION OF THE INSTRUMENT

This Section should include a general description of the instrument, in particular :

- the overall functions of the instrument and the definition of its operational modes during the different phases of the mission
- a summary description of the instrument units and software elements, the functions supported and the redundancy concept adopted
- a list of general instrument constraints (e.g. ground segment reaction time);
- the instrument functional block diagram and a switching diagram showing the location of the telemetry outputs and telecommand inputs.

1.3. SCIENTIFIC OBJECTIVES

This Section should provide a general outline of the scientific objectives of the instrument.

1.4. MISSION PRODUCTS

This Section should provide a description of the scientific data collected by the instrument during the mission lifetime.

2. INSTRUMENT CONFIGURATION AND MODES

2.1. GENERAL

This Section should contain a detailed description of the instrument configuration and modes during the different phases of the mission. A mode (nominal or backup) is defined by a given hardware and/or software configuration, which has to be described for each of the modes identified hereafter. The possible mode transitions (automatic on board or initiated from ground) shall be shown for each phase of the mission.

2.2. INSTRUMENT CONFIGURATION IN LEOP

In this Section it should be indicated whether the instrument needs any particular support (e.g. thermal control) from the spacecraft during the launch and early orbit phase. The Section should also contain a description of the instrument modes and associated configurations (hardware and software) in LEOP.

2.3. INSTRUMENT CONFIGURATION IN ROUTINE PHASE

2.3.1. Nominal Modes

The nominal modes of the instruments during this phase shall be identified.

2.3.2. Back-up Modes

2.3.2.1. Redundant Modes

If the instrument is able to reconfigure some of its units or subsystems to the redundant ones in case of malfunctioning of the nominal ones, it is said that it goes in redundant mode.

The nominal operations of the instrument are maintained without interruption or degradation. Redundant modes, if any, shall be identified in this Section. It shall also be indicated whether the transition from nominal to redundant mode is automatic or must be initiated from on-board or ground command.

2.3.2.2. Survival Modes

A survival mode places the instrument into a safe condition, preventing loss or permanent degradation. The instrument can later be reconfigured from ground to continue observations and achieve a specified performance level. The survival modes also include certain back-up modes triggered by an interruption of certain functions provided by the Service Module (SVM) of the spacecraft, such as power distribution, telemetry and telecommand, attitude control

Survival modes, if any, shall be identified in this section. It shall also be indicated whether the transition mode is automatic or must be initiated from ground command.

3. OPERATIONAL SCENARIO

3.1. GENERAL

This Section should provide a detailed description of the operations to be conducted in the different phases of the mission. It should identify the operational modes applicable to each operation, the mode transitions which are allowed, and the operational constraints at instrument level.

3.2. LEOP OPERATIONS

3.2.1. Description of the Operations

A detailed description of the operations foreseen in LEOP phases shall be given in this Section.

3.2.2. Operational Constraints

3.2.2.1. Planning Constraints

This Section shall provide the planning constraints applicable to the operations discussed. Examples are the need for ground coverage, execution during long contacts, need for real-time interaction with the instruments.

3.2.2.2. Constraints on the Spacecraft

In this Section, the constraints imposed on the spacecraft by the operations in question shall be identified. They may include, for example, special pointing requirements, thermal and power support...

3.2.2.3. Constraints on the Ground Segment

This Section shall identify the constraints that the operations the ground segment. For example, special command procedures, quick operational feedback...

3.2.2.4. Constraints on Other Instruments

The operations may impose particular requirements on other instruments, for example mechanical perturbations below a certain level. Such constraints, if they exist, shall be identified in this Section.

3.2.3. Detailed Timeline of Events

A detailed timeline of the events taking place during the operations shall be given in this Section.

3.2.4. Detailed Operational Procedures

The operational procedures required in this phase of the mission should be defined, including the procedures for mode transitions

Each procedure should be structured as follows :

- an introduction, describing the purpose of the procedure and when it is applicable
- the spacecraft status and resources required to support the procedure and the resulting mode

- the instrument configuration for each step of the procedure

The procedure itself shall contain, for each step, the following information :

- execution constraints
- preliminary verification (i.e. checking before command sending)
- description of the action : command to be sent and telemetry to be monitored to verify the correct execution of the step
- final verification at the end of the step.

3.3. COMMISSIONING AND CALIBRATIONS OPERATIONS

3.3.1. Description of the Operations

This Section shall contain a detailed description of the commissioning and calibration operations and their purposes.

3.3.2. Operational Constraints

3.3.2.1. Planning Constraints

3.3.2.2. Constraints on the Spacecraft

3.3.2.3. Constraints on the Ground Segment

3.3.2.4. Constraints on Other Instruments

3.3.3. Detailed Timeline of Events

3.3.4. Detailed Operational Procedures

3.4. ROUTINE OPERATIONS

3.4.1. Description of the Operations

This Section shall contain a detailed description of the routine operations and their purposes.

3.4.2. Operational Constraints

3.4.2.1. Planning Constraints

3.4.2.2. Constraints or the Spacecraft

3.4.2.3. Constraints on the Ground Segment

3.4.2.4. Constraints on Other Instruments

3.4.3. Detailed Timeline of Events

3.4.4. Detailed Operational Procedures

3.5. COORDINATED OPERATIONS WITH OTHER INSTRUMENTS

3.5.1. Description of the Operations

Any coordinated operation in conjunction with other instruments, such as observation campaigns, shall be presented in detail in this Section.

3.5.2. Operational Constraints

3.5.2.1. Planning Constraints

3.5.2.2. Constraints on the Spacecraft

3.5.2.3. Constraints on the Ground Segment

3.5.2.4. Constraints on Other Instruments

3.5.3. Detailed Timeline of Events

3.5.4. Detailed Operational Procedures

3.6. SPECIAL OPERATIONS

3.6.1. Description of the Operations

Any special operation not discussed in the previous Sections shall be described here.

3.6.2. Operational Constraints

3.6.2.1. Planning Constraints

3.6.2.2. Constraints on the Spacecraft

3.6.2.3. Constraints on the Ground Segment

3.6.2.4. Constraints on Other Instruments

3.6.3. Detailed Operational Procedures

3.6.4. Detailed Timeline of Events

3.7. CONTINGENCY OPERATIONS

3.7.1. Description of the Operations

This section should identify the methods by which the ground segment can identify a failure condition from the analysis of the telemetry data, isolate the source of failure and make the appropriate reconfiguration to put the instrument back into service.

Potential failure cases failure should be identified by means of a systematic failure analysis.

For each failure case, a recovery procedure should be defined and presented in the same format as the nominal operational procedures.

This Section should also identify any expected performance degradation in the instrument in the course of time, and indicate the resultant impact in terms of modified operational constraints.

3.7.2. Operational Constraints

3.7.2.1. Planning Constraints

3.7.2.2. Constraints on the Spacecraft

3.7.2.3. Constraints on the Ground Segment

3.7.2.4. Constraints on Other Instruments

3.7.3. Detailed Operational Procedures

3.7.4. Detailed Timeline of Events

4. GROUND SEGMENT SUPPORT

4.1. SUPPORT DURING FLIGHT OPERATIONS

4.1.1. General

Section 4.1 should define the support needed by the instrument from the ground segment during flight operations.

4.1.2. LEOP

This Section shall identify any specific support needed from the METOP Ground Segment to plan and perform the particular operations in question.

4.1.3. Commissioning and Calibrations

4.1.4. Routine Operations

4.1.5. Coordinated Operations with Other Instruments

4.1.6. Special Operations

4.1.7. Contingency Operations

5. INSTRUMENT DATA BASE

5.1. GENERAL

The instrument Data Base, containing a complete definition of all HK telemetry and telecommand, shall be briefly described in this Section.

The data base itself shall contain sufficient information to allow the design of the flight control software and for setting up the operational telemetry and telecommand data file.

The inputs to the Data Base will be provided through the Telecommand and Telemetry interface data sheets (EXCEL or LOTUS files), as specified in TBD. These inputs will be integrated in the Spacecraft Data Base.

The data base used for AIV will be extracted from the Spacecraft Data Base.

5.2. TELEMETRY SUMMARY

This Section should contain a summary list of the telemetry parameters which belong to the instrument.

5.3. TELECOMMAND SUMMARY

This Section should contain a summary list of the telecommand parameters which belong to the instrument.

6. APPENDIX: FORMAT OF THE INSTRUMENT FLIGHT PROCEDURES

All flight procedures shall be written with the following fixed structure :

1. Purpose
2. Execution Constraints
3. Preliminary Verifications
4. Step Sequence
    - 4.1 Step 1
        - 4.1.1 Purpose
        - 4.1.2 Execution Constraints
        - 4.1.3 Preliminary Verifications
        - 4.1.4 Actions - Verifications
        - 4.1.5 Final Verifications
        - 4.1.6 Final Constraints
    - 4.2 Step 2
    - ...
    - 4.n Step n
5. Final Verifications
6. Final Constraints

APPENDIX II

ELECTRICAL INTERFACES

GENERAL AVIONICS INSTRUMENT INTERFACE CONTROL DOCUMENT (GAICD)

REF. ME-IS-DOR-DOR-PM-0001

[redacted]

Title: **General Avionics Instrument Interface Control Document (GAICD)**

DRD 21

Prepared by:	Team	Date:	15.09.94
Checked by:	F. Tanner		
Product Assurance:			
Project Management:	F. Tanner		20.09.94

Distribution:

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Knoblauch, August	X	EUM	
Lang, Jürgen	X	NOAA	
Langemann, Manfred	X	Spares	5
Schönherr, Klaus	X		
Strauß, Uwe	X		
Tanner, Fred	X		

CHANGE RECORD

Issue	Date	Page	Description of Change	Release
Draft	11.03.94	all	First Issue	21.03.93
Draft B	14.04.94	all	Update for MHS Interface Meeting	14.04.94
Draft C	27.05.94	all	New structure after MTP	27.05.94
			Delete NOAA 2000 part, revise EMC part,	
Draft D	20.07.94	all	Phase A Final Presentation Version	20.07.94
1	15.09.94		Phase A Final Version, see revision marks	20.09.94
		46	table corrected	
		52	clock stability and drift, typos corrected	
		79-84	EMC Design revised	
		86	NOAA conducted emissions approach clarified	
		100	attitude state vector, typos corrected	
		178-180	EMC Performance revised	

CONTENTS

I. TERMS AND DEFINITIONS

Satellite: is composed of the Service Module and the Payload Module

Service Module: provides attitude and orbit control, telecommand and control link with the ground in S-band, satellite monitoring and control, power generation, energy storage and power distribution, interfaces with the launcher

Payload Module: is composed of the PLM Structure including secondary structures like balconies, plinths, etc, the PLM Thermal Control, and the PLM Avionics. The PLM accommodates the majority of Payload elements The PLM Avionics provides the following functions: PLM level command distribution, monitoring and control, electrical distribution of power, signals, and data, Payload Data Handling and Transmission (PDHT) comprising measurement data acquisition from instruments, multiplexing, encryption, and storage, measurement data transmission to ground directly in X-band and continuous transmission of LRPT and HRPT formatted data-streams.

Payload: is the assembly of the Instruments to be embarked on METOP.
The Payload falls into the following groups:

- NOAA KLM instruments (KLM)
- European Meteorological Instruments (EMI) like IASI and MHS
- ESA Developed Instruments (EDI) like ASCAT and MIMR
- Announcement of Opportunity instruments (AO) like ScaRaB and GOME

II. ABBREVIATIONS

AIT	Assembly, Integration & Test
AIV	Assembly, Integration & Verification
AMSUA1	Advanced Microwave Sounding Unit 1
AMSUA2	Advanced Microwave Sounding Unit 2
AO	Announcement of Opportunity
AOCS	Attitude and Orbit Control System
AR5	Ariane 5 Launcher
ASCAT	Advanced Scatterometer
AVHRR/3	Advanced Very High Resolution Radiometer
BER	Bit Error Rate
BOL	Beginning of Life
BPSK	Binary Phase Shift Keying
CMD	Command
DCS/2	Data Collection System
EGSE	Electrical Ground Support Equipment
EIRP	Equivalent Isotropic Radiated Power
EM	Engineering Model
EMI	European Meteorological Instruments, i.e. IASI and MHS
EMC	Electromagnetic Compatibility
ENVISAT	Environmetal Satellite
FM	Flight Model
FMU	Formatting and Multiplexing Unit
FOV	Field of View
GOME	Global Ozone Monitoring Equipment
G/T	Gain over temperature
GS	Ground Segment
HIRS/3	High Resolution Infra Red Sounder
HK	House Keeping
HRPT	High Resolution Picture Transmission
I/F	Interface
IASI	Infrared Atmospheric Sounding Interferometer
ICD	Interface Control Document
ICU	Instrument Control Unit
KLM	NOAA K, L, M series instruments to be embarked on Tiros K,L,M, and METOP

LEOP	Launch and Early Orbit Phase
LRPT	Low Resolution Picture Transmission
MCMD	Macro Command
MGSE	Mechanical Ground support equipment
MHS	Microwave Humidity Sounder
MIMR	Advanced Microwave Imaging Radiometer
MLI	Multilayer Insulation
MTBF	Mean Time Between Failures
MTBR	Mean Time Between Repairs
N/A	Not Applicable
NIU	NOAA KLM Instruments Interface Unit
NOAA	(US) National Oceanic & Atmospheric Administration
OBDH	Onboard Data Handling System
P/L	Payload
PCU	Power Conditioning Unit
PDHT	Payload Measurement Data Handling and Transmission
PDU	Power Distribution Unit
PLM	Payload Module
QPSK	Quadri Phase Shift Keying
RAU	Remote Adaptation Unit
RFC	Radio Frequency Compatibility
RFI	RF Interference
RTU	Remote Terminal Unit
S/S	Subsystem
S/W	Software
ScaRaB	Scanner for Radiation Budget
SCOE	Special Checkout Equipment
SVM	Service Module
TBC	To be confirmed
TBD	To be defined
TBR	To be reviewed
TBS	To be specified
UHF	Ultra-High Frequency
VCDU	Virtual Channel Data Unit

III. DOCUMENTATION

Reference Documents

The following documents are for reference only. They can be made applicable to the extent explicitely specified in this document.

Ref.	Doc. No.	Title	Issue	Date
AD2	ME-RS-ESA-SY-0001	Space Segment Requirement Specification	1 R 1	Apr. 92
ID1	IS-3267415	ATN-KLM General I/F Specification	Rev. C	18.10.91
ID2	GE-IS-2617547	Unique Instrument Interface Specification for AMSU-A1	Rev. N	16.03.92
ID3	GE-IS-2624483	Unique Instrument Interface Specification for AMSU-A2	Rev. L	16.03.92
ID4	GE-IS-20029950	Unique Instrument Interface Specification for AVHRR/3	Rev. 0	20.01.92
ID5	GE-IS2285780	Unique Instrument Interface Specification for HIRS/Z	Rev. G	27.10.91
ID6	GE-IS-3267402	Unique Instrument Interface Specification for DCS	Rev. A	17.07.92
RD10	PO-ID-DOR-PL-0026	ENVISAT Common Interface Control Doc.	1 Rev B	20.01.94
RD10a	SPE 1211368 015	ENVISAT Avionics Interface Specification	2 Rev B	Feb. 94
ID11	IA-TN-1.0.1.-61-IPT	IASI Instrument Data List	1	15.06.93
ID12	PO TN RAL AT 0003	AATSR Instrument Data List	12	07.06.93
ID14	PO TN ESA SCA 0072	SCaRaB Instrument Data List	draft B	17.11.92
ID15	PO RS ALS MI 1009	ICD MIMR	1	20.05.93
ID16	PO LI DOS SC 1170	ASCAT Data List	2	22.04.92
ID17	EPS/MHS/ICD/9301	MHS Instrument Data List	1	04.10.93
ID25	PO-RS-DOR-PL-0002	ENVISAT-1 Electrical Requirement Specification	Rev. C	25.10.93
ID26	PO-RS-DOR-PL-0006	EMC Requirement Specification		
ID27	PO-RS-DOR-PM-0010	OBDH & Measurement Data Protocol Specification	1 Rev B	
ID28	PO-RS-DOR-PL-0007	Command & Control Specification	2 Rev A	

IV. INTRODUCTION

This document specifies the common avionics interfaces for instruments on the METOP-1 satellite. Avionics interfaces comprise:

- All electrical interfaces between the instruments and the satellite
- All command and control interfaces including instrument operation
- EMC / RFC issues
- EGSE interfaces between instruments and instrument EGSEs to the PLM/Satellite EGSE
- Ground operation of instruments as concerns electrical and functional testing

METOP instruments can be classified into two categories, i.e.

- NOAA KLM instrument
- European Meteorological Instruments, EDI and A.O. instruments.

Different than for thermal and structure where common requirements do apply equally to all instruments, the avionic interfaces of above catagories differ significantly.
One can even say that the different avionics interfaces lead to the split into the above categories.

To support the different categories of interfaces in an efficient manner, avoiding as far as possible duplication of requirements, the General Instrument Interface Control Document for the METOP instruments has been separated into two separate parts, i.e.

- a General Instrument Interface Control Document (GIICD)
- a General Avionics Instrument Interface Control Document. (GAICD)

This General Avionics Instrument ICD comprises two main sections, i.e.

A) KLM Instruments Interface Characteristics
B) EMI, EDI & A.O. Instruments Interface Characteristics

Both parts define the general requirements for both sides of the interface,

- the instrument
- the satellite.

This general avionics ICD will be complemented, precised or overwritten where necessary by the dedicated

- instrument specific avionics interface control document (ISAICD)

which will be established for each instrument at a later stage.

This issue of the general avionics instrument interface control document has been prepared for the Final Presentation Data Package of the METOP-1 Phase A study.
As concerns the interfaces to the NOAA KLM instruments actions for better understanding of the interfaces are still pending and could consequently not be reflected in this issue, this is especially true for the instrument operation and power switching. On the other side the electrical interfaces are available from the NOAA instrument unique ICD's and are reflected here even if in many cases the level of detail is exceeding Phase A level. Similar applies for part B where electrical interfaces have been defined along the ENVISAT design and are also relatively detailed. It has to be understood that these sections will have to be reviewed in later phases and will be subject to changes.

From the above a certain imbalance of the various chapters of this document at this stage could not be avoided without suppressing available detailed information, which was not considered appropriate for an early iteration process with all instrument suppliers.

A. KLM INSTRUMENTS INTERFACE CHARACTERISTICS

A.1 Introduction

KLM instruments interfaces are described for the TIROS mission in ID 1 to ID 6. The general approach for the METOP mission in this respect shall be to simulate the TIROS interface as is necessary for the instruments to provide their proper performance. Interface characteristics available on TIROS but not needed by the instruments for their proper operation should not be simulated on METOP. Therefore differences to TIROS do not a priori indicate a wrong understanding but should be reviewed on their compatibility with the existing instrument design. In describing the various interfaces in this part A, clarification with EUMETSAT and NOAA on the interfaces shall be started to support an early agreement between the relevant parties.

A.2 Overview

The avionics interface between the METOP Payload Module (PLM) and the NOAA KLM instruments is mainly handled via the Power Conditioning Unit (PCU) and the NOAA Interface UNIT (NIU). For adaptation of the single ended interfaces of NOAA instruments to the distributed star point grounding concept an additional Remote Adaptation Unit (RAU) has been implemented for those KLM instruments which have significant harness lengths to the NIU.

Figure A.2-1 shows the METOP PLM Blockdiagram whereas Figure A.2-2 gives an overview on the various electrical interfaces between the PLM and the KLM instruments.

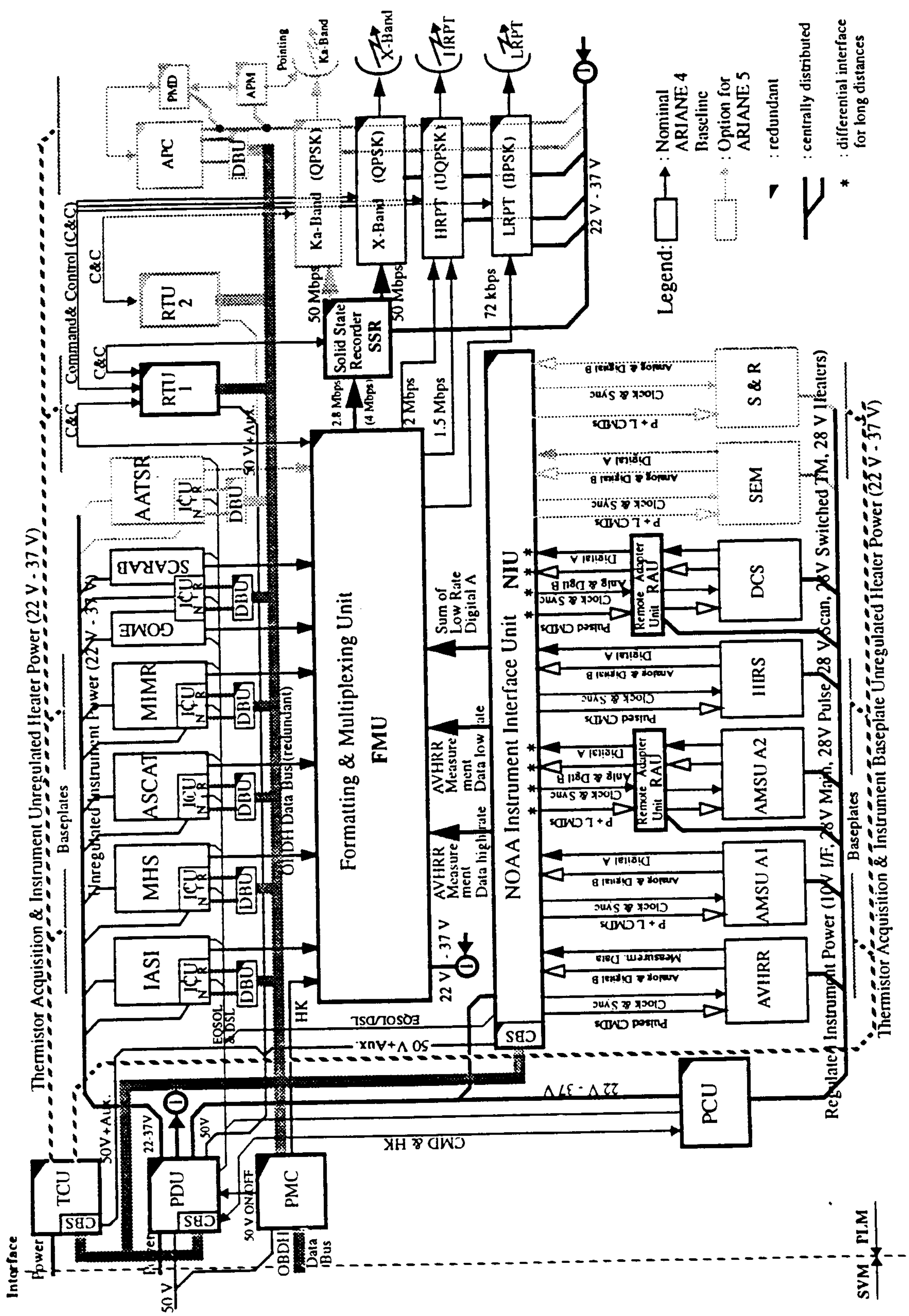


Fig. A.2-1: Overall PLM Electrical Block Diagram

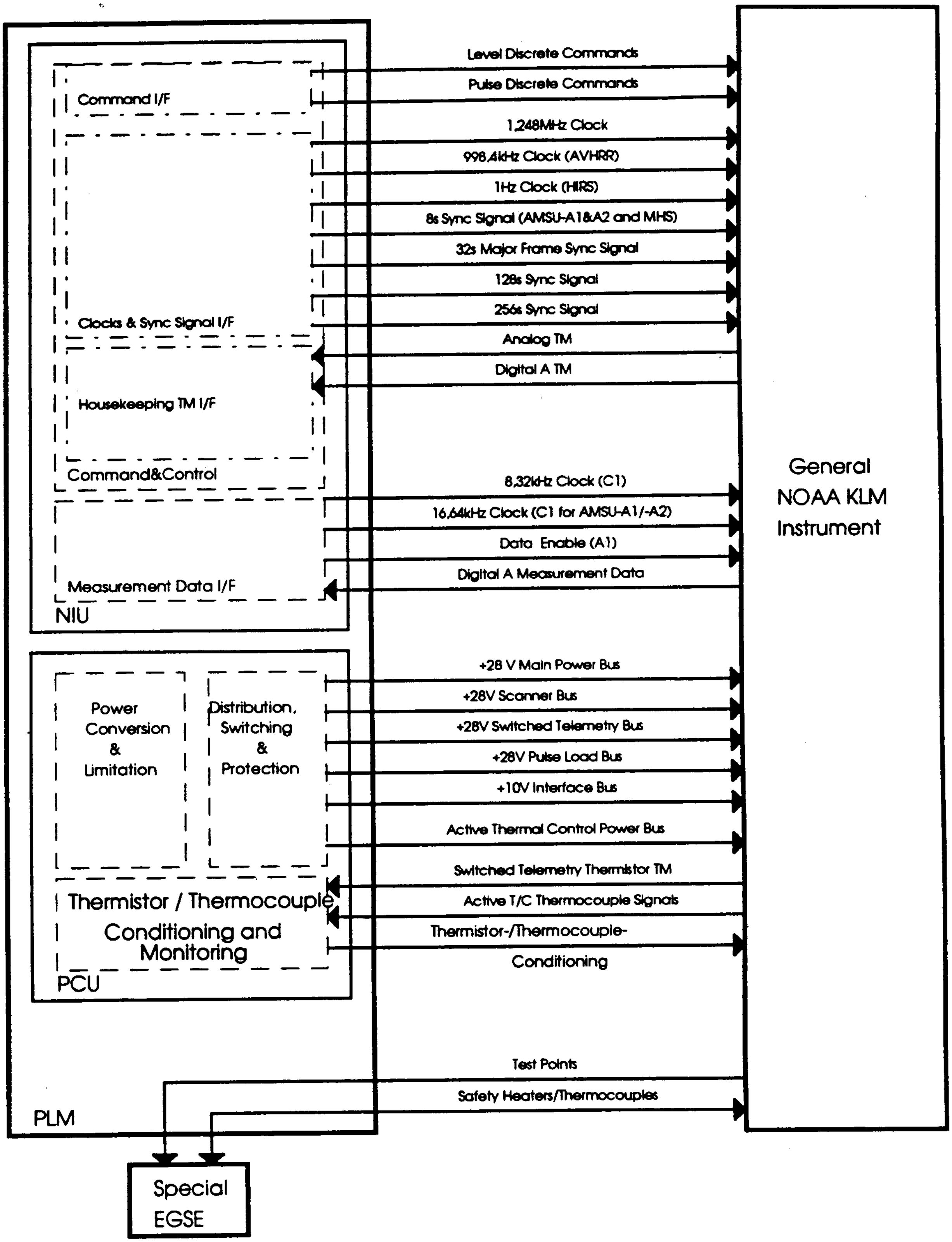


Figure A.2-2: PLM to KLM Instrument Electrical Interfaces Overview

A.3 Electrical Power Interfaces

The NOAA KLM type instruments doe require regulated power for operation and thermal control. Therefore these instruments will be provided with up to 6 types of power interfaces by the PLM.

- A regulated +28 Volt Main Bus with high power quality as primary source for the instruments
- A regulated +28 Volt Scanner Bus with high power quality as power source for elements like scanners which need to be powered during launch and nominal operation as well as in case of non-nominal pointing of the satellite, e.g. in safe mode, respectively
- a regulated +28 Volt Pulse Load Bus for powering stepper motors, heaters, and other high current pulse loads which cannot meet the Main Bus current ripple specifications.
- A regulated +28 Volt Switched Telemetry Bus for monitoring of the instrument temperatures when the instrument is off.
- A regulated +28 Volt Active Thermal Control Bus for instrument thermal control during periods of nominal operation, operated from the PLM thermal control on TBD thermistor / thermocouples readings.
- A regulated +10 Volt Interface Bus for powering of standard control interfaces in the instruments

All these power busses will be generated inside the internally redundant Power Conversion Unit (PCU) and will be individually switched and protected per instrument.
The availibility of these busses in the various PLM modes is shown in table A.3-1.
The returns of all power busses will be grounded to satellite ground inside the PCU.

Modes	Inert	Ground	LEOP	Stdby	Operable		Fix	Safe
Power					Inst. Op.	Inst. Off		
Main Bus	off	any	off	on	on	off	off	off
Scanner Bus	off	any	on	on	on	off	off	on
Pulse Load Bus	off	any	off->on	on	on	on	on	on
Switched TM Bus	off	any	off->on	on	on	on	off	off
Active. T/C Bus	off	any	off	on	on	off	off	off
10 V I/F Bus	off	any	off->on	on	on	off	off	off

off->on = switched on by ground command after successful solar array deployment and when necessary from thermal point of view

Table A.3.-1: Power Bus Availibility

A.4 Command and Control

The Payload Module Controller (PMC) shall be the master controller for Command & Control operations of the NOAA KLM instruments.
The NIU shall translate/convert the PMC functional and electrical interface into NOAA Instrument Interface compatible structures and signals.

A.4.1 Protocol

The PMC shall use the low level protocol (ref. to ID27 for the CBS) to communicate with the NIU via the OBDH Data Bus. The NIU shall transfer all telecommands received from the PMC for instrument commanding into the required low level commands specified in sect. A.4.2.
Additional low level satellite commands in contingencies shall be acquired from the NIU which are the Equipment-Switch-Off-Line (EQ-SOL) and the Depointing Signal Line (DSL). The NIU shall translate these signals into TBD commands for appropriate switch-down of the instrument. This TBD command sequence shall be defined conform to the relevant switching of power-buses by the PCU.
For commands issued from the PMC in the low level protocol for the interrogation of HK-TM parameters, the NIU shall acquire the respective HK-TM from the instrument and report it on the response bus.

A.4.2 Telecommands

The Satellite Payload Module Controller (PMC) of the PLM Avionics shall derive low level commands for KLM instruments from KLM macrocommands expanded within the PMC's KLM application software.
The PMC shall distribute these commands over the OBDH Data Bus to the standard bus coupler (CBS) inside the dedicated NOAA Instrument Interface Unit (NIU). The NIU shall transfer these low level commands into the relevant telecommands defined for the instrument.

The PMC shall be able to store the commands for NOAA instruments as time tagged commands in the MCMD Queue, which is to be loaded or to be updated from ground. Preprogramming for up to 36 hours shall be provided.
The execution uncertainty for MCMDs by the PMC shall be < 125 ms (TBC).
The storage capacity for KLM instruments time tagged commands shall be TBD words for max. 36 hours of operation.

The PMC shall monitor correct execution of commands and perform autonomously TBD corrective measures in case of monitored anomalies.

The NIU shall provide two types of commands:

- **Level Discrete Commands**, representing ON or TRUE condition to the instrument full time until another command is given to change the state to OFF or FALSE.
- **Pulse Discrete Commands**

Level Discrete Commands:
The PMC/NIU shall be able to combine a series of level discrete commands in a coded fashion to a parallel **binary code of bilevel signals.** To change the status of this code a sequence of Low Discrete commands shall to be sent with a time delay in the order of max. 1s between successful line changes.
The NIU shall be able to change multiple bits (up to 8 bits of one parallel output buffer, TBC) with one single command.

Note:
In order to eliminate the "state stepping" of a command sequence for achieving the final desired state an additional Level Discrete or a Pulse Discrete command line could be used from the instrument as "strobe" signal, i.e. the combined series of level discrete commands is valid to the instrument with the occurrence of this strobe.

Pulse Discrete Commands:
The ON or TRUE condition shall be issued in form of a pulse on a single line. This type of command shall normally be used for latching relays. The pulse ON-state shall have a duration of 60 ± 5 ms for all instruments

A.4.3 Housekeeping Telemetry

The PMC shall acquire NOAA instrument HK telemetry from the NIU cyclically. PMC HK acquisition shall be performed in form of low level acquisition interrogations to the NIU. On reception of a NOAA HK acquisition command from the PMC the NIU shall acquire the actual value of the interrogated HK parameter from the instrument and deliver it as part of a valid response to the PMC via the OBDH data bus.

All analog and Digital B data of the instruments shall be acquired by the NIU.
Analog signals shall be acquired and converted to 8 bit words with 20 mV resolution.
Digital B data shall be acquired by the NIU individually and assembled to 8-bit words.

The Digital B data shall be acquired once per 16 s sampling interval by the interrogations timeline of the PMC.
The PMC shall send the acquired NOAA KLM instrument HK data to the FMU for real-time downlink every 8 (TBC) s.

A +28V Switched Telemetry Bus (see sect A.6.2) shall be supplied to each instrument. This bus shall power one or two thermistors for instrument temperature monitoring when the instrument is "OFF". The thermistor conditioning (half-bridge TBC) shall be performed by the NIU (TBC). The thermistor signals shall be acquired from the instrument during "Instrument-Off" periods according to PMC interrogation.The conditioned thermistor signal shall be acquired and reported from the NIU as the respective response on the Response bus.

A.4.4 Satellite Services

The following satellite services shall be provided by the PMC/NIU for the NOAA instruments.

A.4.4.1 Housekeeping Formats

The PMC shall assemble NOAA instrument HK data to format data sets similar to ICU formats which shall build the Source Data of the CCSDS Source Packets. The PMC shall perform complete Source Packet generation including the datation in the secondary header of the NOAA instrument HK data conform to the layout of fig. A.4.4.1-1.

Packet Identification				Packet Sequ. Ctl.		Packet Length	Source Data Field		Packet Err. Ctl. (CRC)
Version No.	Type	Secondary Header Flag	Applic. Process ID	Sequ. Flags	Source Sequ. Count		Secon. Header (Time)	Source Data	
3	1	1	11	2	14	16	48	variable	16
2 octets				2 octets		2 octets	6 octets	Var.	2 octets

Fig. A.4.4.1-1 : Source Packet Layout

A.4.4.2 On-Board Monitoring and Corrective Action

The PMC shall monitor TBD instrument digital HK-TM parameters on instrument status and TBD analog instrument parameters for out-of limit checks.
In case of detection of an anomaly in the monitored instrument status and/or out of bounds detection of an analog parameter the PMC shall perform TBD corrective action. The corrective action shall be performed only after two consecutive failing HK samples of the same parameter (filtering of two, refer to ID 28).
Dedicated History Reporting shall be performed by the PMC concerning any autonomous action taken. History files shall include the HK TM which has been identified as anomaly causing the corrective action taken. The PMC shall also format the History reports into Source Packets.

A.4.4.3 Orbit State Vector

The Orbit State Vector shall be sent in TBD format from the PMC to the FMU every 1 s (TBC).
The PMC shall generate a dedicated Source Packet per 1 s time interval containing:

- Orbit State Vector
- Attitude State Vector
- on-board time

and distribute it to the FMU.

A.4.4.4 Attitude State Vector

The Attitude State Vector shall be sent in TBD format from the PMC to the FMU every 1 s (TBC). The PMC shall generate a dedicated Source Packet per 1 s time interval containing:

- Orbit State Vector
- Attitude State Vector
- on-board time

and distribute it to the FMU

A.4.4.5 Auxiliary Data

Any Auxiliary data which is to be merged into the NOAA K, L, M data stream shall be formatted and packetized from the PMC into Source Packets and distributed to the FMU.
This Auxiliary data may be Administration Messages to the NOAA user community and TBD others.
One of these is proposed as the correlation information between satellite time (in CUC) and ground UTC of the latest actual time correlation (TBC) (ref. to sect. A.4.4.7).

A.4.4.6 Synchronisation

Clocks:
The instruments shall be operated synchronously with reference to the following general clock signals

- master reference (for el. characteristics ref. to sect. A.6.3)
- 1 Hz clock, derived from the master clock, (for el. characteristics ref. to sect. A.6.3)

One exception for the AVHRR instrument solely shall be the distribution of

- a 0.9984 MHz clock instead of the 1.248 MHz master reference (for el. characteristics ref. to sect. A.6.3)

Synchronization Signals:
Synchronization signals shall be generated for instrument operations to synchronize for data collection activities of Digital A data (measurement data stream). For this item refer to sect. A.5.
For instrument operations to synchronize for data collection activities (with reference to the TIROS Information Processor (TIP)) the following pulses shall be distributed to the instruments (to be harmonized with sect 6.4):

- a major frame sync pulse, every 32-seconds
- a 128-second calibration pulse
- and a 256-second calibration pulse

The calibration pulses shall initiate the scanline counts for instruments with stepper mirrors.

For instrument operations to synchronize for data collection activities with reference to the TIROS AMSU Information Processor (AIP) the following pulses shall be distributed to the instruments:

- an 8-second sync pulse for AMSU-A1 to synchronize the instrument output data format with the start of each AIP (on TIROS) frame.

A.4.4.7 Datation

The NIU shall perform all necessary datation of measurement data (HK-TM data excluded) and include it in the secondary header (TBC) of the individual Source Packets which are to be generated from the NIU.
The resolution and the uncertainty of the time information shall be TBD (1ms resolution is assumed).
The time generation function shall maintain the time in CCSDS Unsegmented Time Code (CUC).
The time generation function shall synchronize the NOAA K, L, M instrument time with the satellite time on request.
It is proposed that the GS will uplink the actual correlation information between satellite time in CUC and ground UTC in TBD time intervals, e.g. every 24 or 36 hours
The CUC to UTC time correlation information shall be inserted in the auxiliary data source packet.

A.4.5 Command & Housekeeping Data Budget

Instrument	Pulse Commands (60 ± 5 ms)	Level Commands	Analog Housekeeping TM	Switched Telemetry Bus	Digital Housekeeping TM (B)
AVHRR	30	0	22	0	15
HIRS	26	0	14	2	14
AMSU A1	4	10	21	6	12
AMSU A2	4	9	11	4	10
DCS-2 **	24	0	9	4	15
*SEM**	*7*	*6*	*9*	*4*	*9*
SARR **	*14*	*18*	*13*	*9*	*4*
SARP-2 **	*22*	*0*	*9*	*3*	*13*

* Please note that SAR and SEM are only listed optional in *Italic letters* as they do not belong to the baseline payload
** SAR and DCS inputs from lasted presentation are TBC because they deviate from the Unique Interface Specifications

A.5 Measurement Data

METOP measurement data services, transmitted to ground through the LRPT, HRPT and Global Data downlinks, will be formatted according the CCSDS Data System Standards. CCSDS Formatting features in general two services: Source Packet formatting and Transfer Frame formatting.

Source Packet formatting, as expressed by the term, is normally performed by the instrument, whereas Transfer Frame formatting is a service provided by the Payload Data Handling (PDHT) System.

Since the NOAA K, L, M instruments deliver their measurement data in a form compliant to the TIROS-N Data Handling system, the METOP PDHT has to provide the Source Packet service to each KLM instrument. In order to keep the KLM instrument interfaces unchanged w.r.t. the TIROS implementation, the METOP PDHT has to provide the same data acquisition procedure as well as the electrical I/F characteristics to the instruments as the TIROS-N Data Handling system.

All measurement data acquisition from the NOAA KLM instruments on METOP will be performed by the NOAA Interface Unit (NIU).

Instrument data of one complete scan cycle shall be assembled into one source packet of each instrument.

The measurement data shall be acquired from the NOAA KLM instruments via the standard Digital A interface. The Digital A interface is a serial interface and consists of the following three single lines:

- <u>The Select Interval (A1)</u> from the NIU to the instrument enables the data transfer of one or several bytes from the selected instrument.
- <u>Data Clock (C1)</u> from the NIU to the instrument provides the synchronisation signal for the serial data transfer.
- <u>Data Line (D1)</u> from the instrument to the NIU transfers the data synchronised to the clock (C1) signal

The clock signal (C1) is continuously available to all Digital A users at a 8.32 kHz or 16.64 kHz rate. The select interval (A1) is enabled only when a particular Digital A source is to be interrogated and stays in logic ø (high) when not selected. When interrogated, 8 bits (one byte) or a series of bytes of data are serially shifted from the instrument to the NIU in synchronism with the clock (C1).

<u>Timing:</u> The timing of the NIU/Digital A interface is shown in Figures A.5.1a&b. The first (most significant) bit of information to be transferred shall be established; i.e., reached 90 % of value, a maximum of 48 μs before the logic "0" to logic "1" transition of the "C_1" shift pulse signal occurs or, as a minimum, coincident with the logic "0" to logic "1" transition of the first "C_1" shift pulse signal. The NIU clocks (accepts) the data on the logic "1" to logic "0" transition of the "C_1" shift pulse.

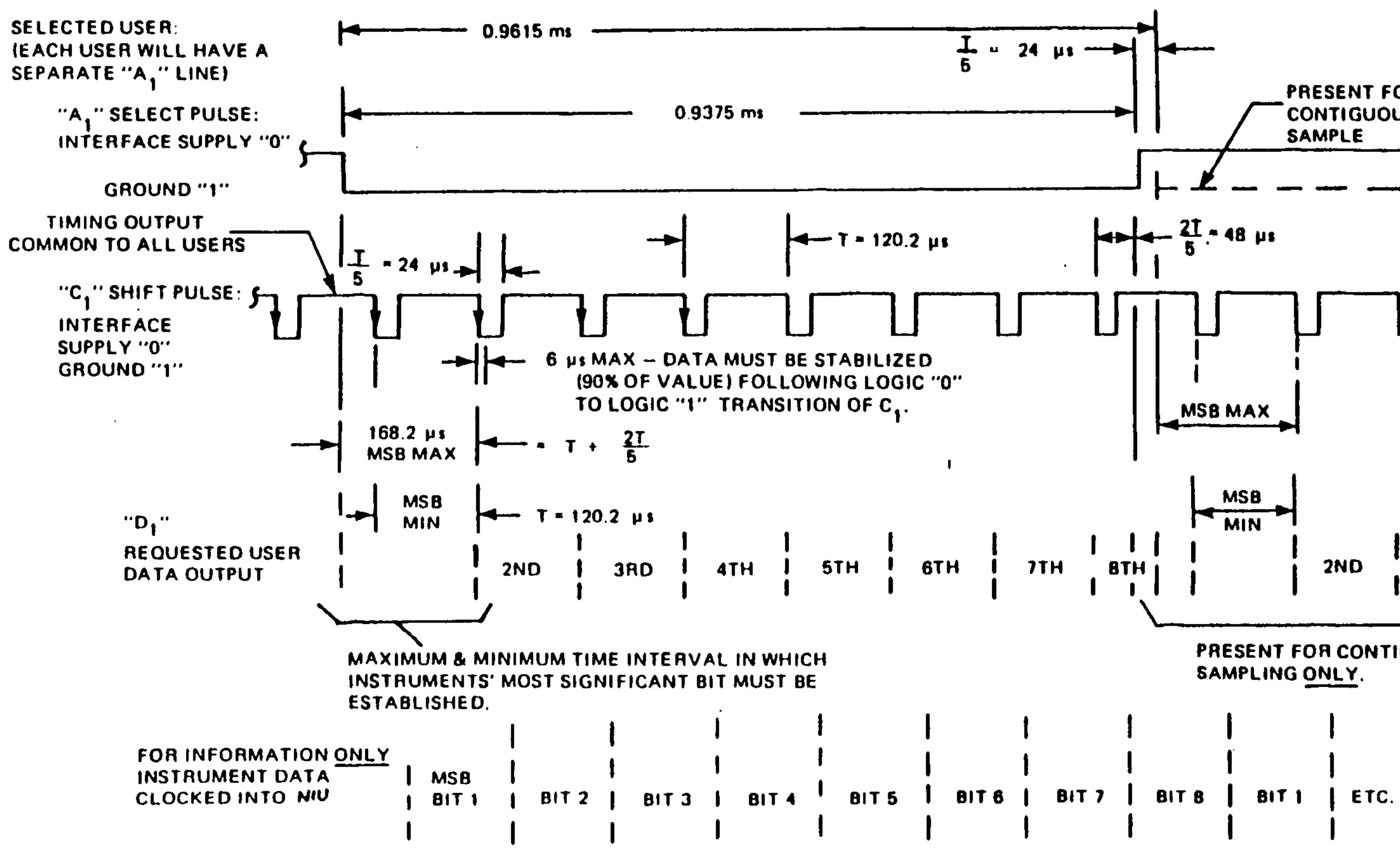
The second through eight most significant bits shall be established by the logic "0" to logic "1" transition of the remaining seven "C_1" shift pulses. The time delay, from the point at which the above transition is 70 % complete to the point at which the transition, if any, of the new data bit being shifted into the instrument output buffer is 90 % complete, shall be less than 6.0 µs. These bits shall remain at their levels until the logic "0" to logic "1" transition of the next "C_1" shift pulse. The NIU clocks the data on the logic "1" to logic "0" transition of the "C_1" shift pulse.

The 6 µs settling time shall be measured at the instrument data output connector with the following loads on the output data line:

CMOS Users:	1000 pf to interface ground

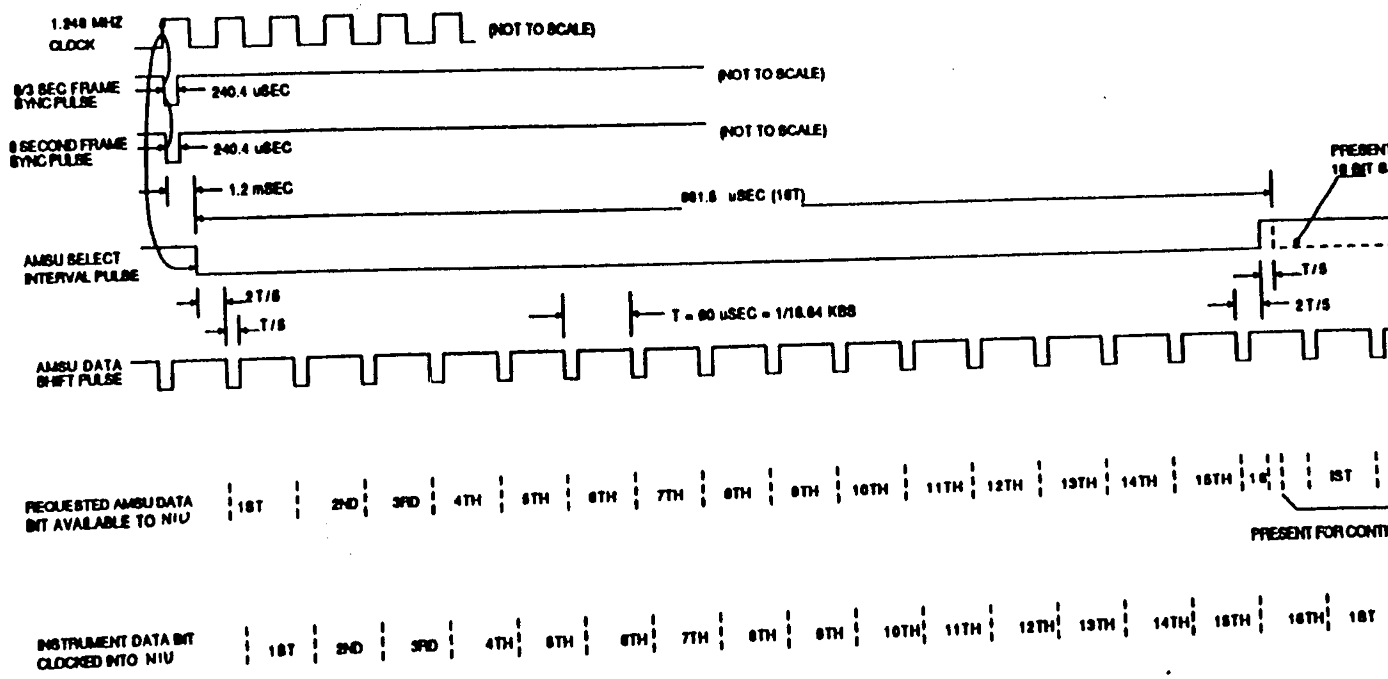
<u>Contiguous Sampling of the Same Data Source:</u> If "n" 8-bit (16 bit) samples of Digital A input occur consecutively in a sampling sequence, the "A_1" pulse generated by the NIU shall be "n" sample intervals wide with a 24 µ sec transition pulse occurring between each 8-bit (16-bit) sampling interval as shown in Figure A.5-1.
The phasing of the interface signals is described in the Digital A Interface timing diagram A.5-1a and A.5-1b. The Digital A interface shall have the electrical characteristics described in chapter A.6.4.

Figure A.5-1a: NIU/Digital A interface (8320 bps)



NOTES:
- LOGIC "1" (ACTIVE LEVEL) = GROUND
- INSTRUMENT DIGITAL "A" DATA OUTPUT INTERFACE TO BE ACTIVE ONLY DURING "A_1" SELECT INTERVAL.
- TIROS-N STANDARD FAST INTERFACE WILL BE USED FOR TRANSFER OF ALL DATA AND CONTROL SIGNALS.
- GROUND REFERENCED TO INTERFACE GROUND.

Figure A.5-1b: NIU/Digital A Interface (16 640 bps)



NOTES:
(1) LOGIC 1 (ACTIVE LEVEL) = GROUND.
(2) INSTRUMENT DATA OUTPUT INTERFACE TO BE ACTIVE ONLY DURING SELECT INTERVAL PERIOD.
(3) TIROS-N STANDARD FAST INTERFACE WILL BE USED FOR TRANSFER OF ALL DATA AND CONTROL SIGNALS.
(4) GROUND REFERENCED TO INTERFACE GROUND.
(5) AMSU SELECT AND SHIFT PULSES ARE CLOCKED BY LOW TO HIGH TRANSITION OF THE 1.248MHz CLOCK AND BY THE HIGH TO LOW TRANSITION OF THE 8 SECOND SYNC PULSE.
(6) DELAY FROM THE 8 SEC SYNC PULSE TO THE AMSU SELECT INTERVAL PULSE IS 1.23MSEC +/- 30USEC FOR A1, 13.73MSEC +/- 30USEC FOR A2, AND 20.43MSEC +/- 30USEC FOR B.

A.6 ELECTRICAL INTERFACES

A.6.1 Electrical Interfaces Overview

The overall PLM electrical architecture is shown in Figure A.6.1-1.

The various types of electrical interfaces provided by the PLM for NOAA-KLM instruments are shown in table A.6.1-1 and depicted in figure A.6.1-2. The shown values are preliminary, instrument specific and defined in detail in the Instrument Specific Avionics ICD (ISAICD).

Electrical Interface	Number of Interfaces						
	AVHRR	AMSUA1	AMSUA2	HIRS	DCS	*SEM*	*S&R*
Power I/F´s							
+28V Main (Power) Bus	1	1	1	1	1	*1*	*1*
+28V Scanner Bus	1						
+28V Switched Telemetry Bus		1	1	1	1	*1*	*1*
+28V Pulse Load Bus		1	1	2			
+ 28 V Active T/C Heater	1			1		*TBD*	*2*
+10V Interface Bus	1	1	1	1	1	*1*	*1*
Command & Control I/F´s							
Level Discrete Commands		10	9			*6*	*18*
Pulse Discrete Commands	30	4	4	26	24	*7*	*36*
1,248MHz Clock		1	1	1		*1*	
998,4kHz Clock	1						
1Hz Clock				1			
8s Sync Signal		1	1				
32s Major Frame Sync Signal				1	1	*1*	
128/256s Calibration Signal				256s			
Analog Housekeeping TM	22	21	11	14	9	*9*	*22*
Digital HK TM (Digital B)	15	12	10	14	15	*9*	*17*
Switched TM Thermistor TM		6	4	2	4	*4*	*12*
Active T/C Thermocouples	1			1		*2*	*2*
Survival Heaters On/Off Cmds		1	1				
Measurement Date I/F´s							
8,320kHz Clock	ISAICD			1	1	*1*	
16,64kHz Clock	ISAICD	1	1				
Data Enable Signal	ISAICD	1	1	1	1	*1*	
Digital A Data Interface	ISAICD	1	1	1	1	*1*	
Test I/F´s							
+ 28 V Safety Test Heaters	TBD	TBD	TBD	TBD	TBD	*TBD*	*TBD*
Test Points	TBD	TBD	TBD	TBD	TBD	*TBD*	*TBD*

Table A.6.1-1: Available/Used Electrical Interfaces

Dornier GmbH		Doc.No.: ME-IS-DOR-PM-0001
Project: **METOP-1**		Issue: Draft D Date: 20.07.94
		Sheet: 31

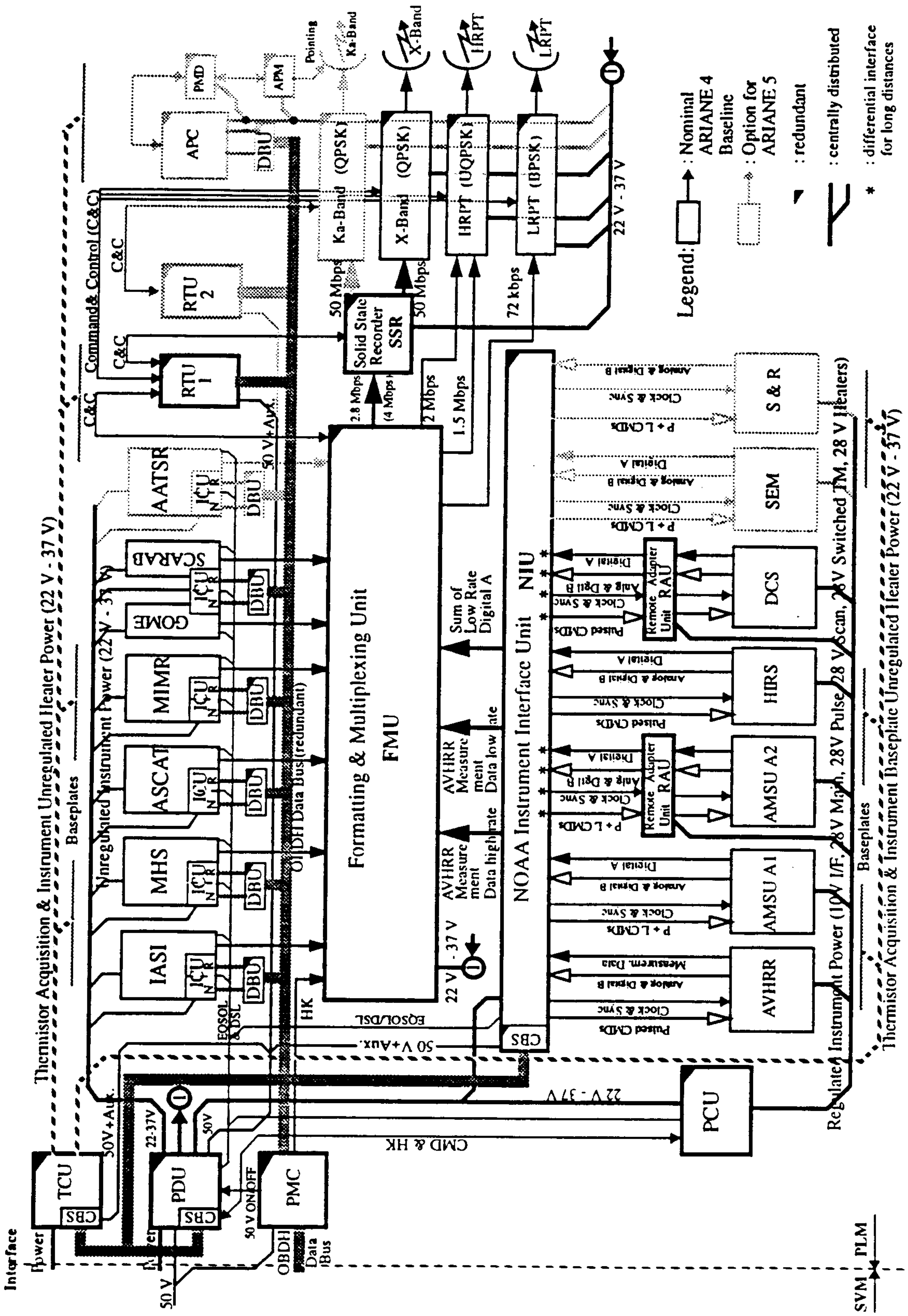


Figure A.6.1-1: Overall PLM Electrical Block Diagram

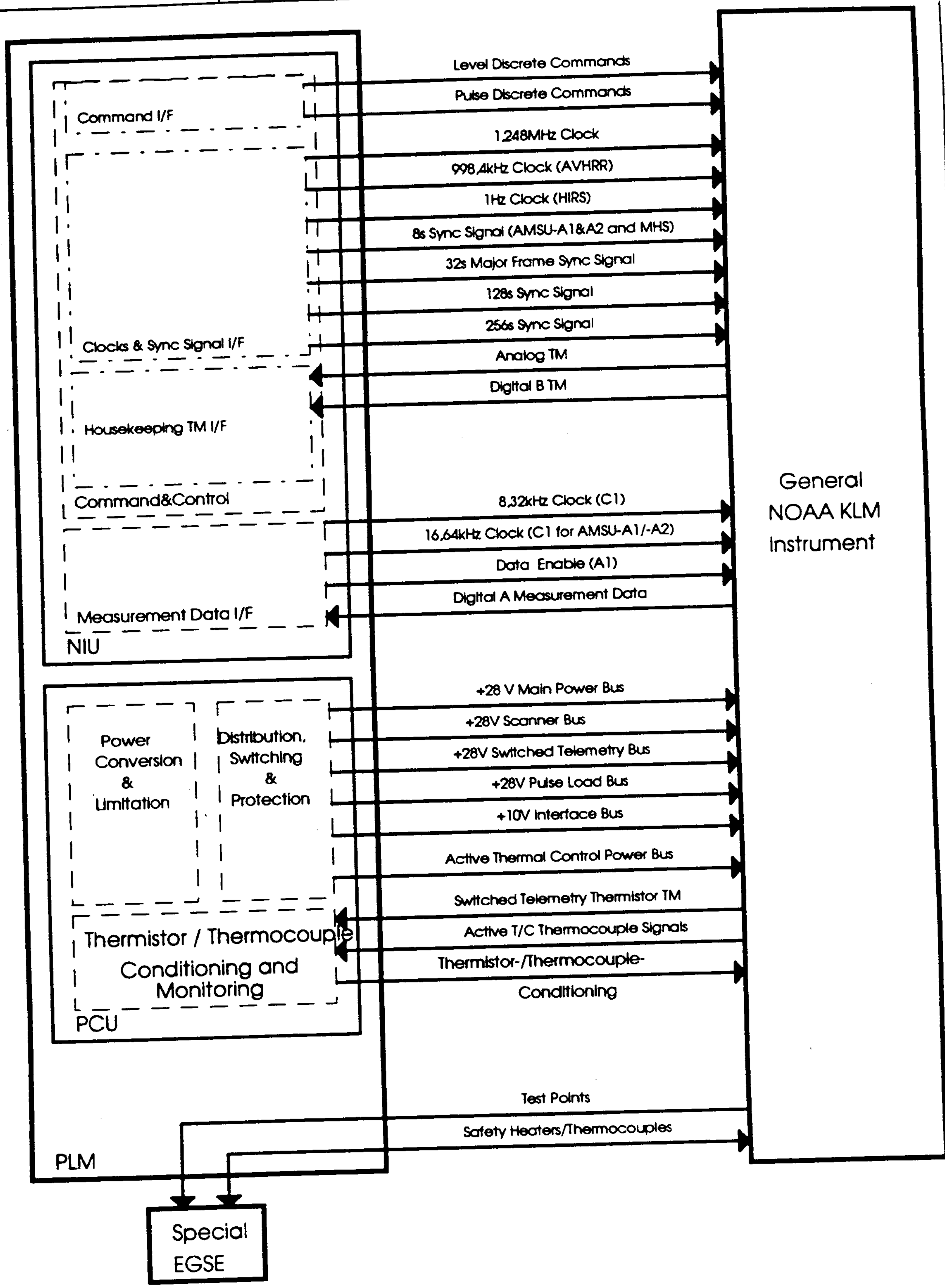


Figure A.6.1-2: PLM to KLM Instrument Electrical Interfaces Overview (RAU not shown)

A.6.2 Power Interfaces

A.6.2.1 Power Demand

The individual instrument power demands for BOL & EOL during all modes, and the required outlet dimensions are defined in the Instrument Specific Avionics ICD (ISAICD). Table A 6.2.1-1 provides a summary of the actual power consumption.

KLM Instruments	Power in W, nominal Mode with recording	remark
AVHRR	29	
HIRS	25	
AMSU-A1	97	
AMSU-A2	41	
DCS	30	on SVM
KLM total	222	

Table A.6. 2.1-1. Preliminary Power Consumption Overview

A.6.2.2 + 28 Volt Main (Power) Bus

A.6.2.2.1 Electrical Interface Description

A.6.2.2.1.1. PLM Side

The + 28 Volt Main Bus is a regulated bus, supplied to each instrument. This bus is classified as a well regulated bus, with minimum ripple and transients, and shall be considered the primary source of power for each instrument.
It is provided from the PLM by conversion and regulation of the 28 V unregulated power.
The Main Bus I/F circuit is shown in figure A.6.2.2.1-1.
The Main Bus power line is protected against short circuit by fuses and switched on the PLM side. Except for fault conditions or spacecraft emergency, the PLM switch shall not be used to switch the instrument rated power.
The PCU provides redundant Main Power sources and fusing which will be selected on PLM side. Instruments will have a single Main Power connection utilising a single Main Power line.
Power line return is connected to the electrical ground on the PLM side.
The various Main Bus power outlets are protected by fuses to a maximal current as specified in the relevant ISAICD.

A.6.2.2.1.2. Instrument Side

See description in the relevant ISAICD.

A.6.2.2.2 Connectors

A.6.2.2.2.1. PLM Side

The connector type used for the Main Bus lines on PLM side is TBD.
The compatibility with the instrument side will be ensured by the PLM.

A.6.2.2.2.2. Instrument Side

The connector type used for the Main Bus lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for connection of several/all power buses.

A.6.2.2.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

T B D

Figure A.6.2.2.1-1: Circuit Diagram - Main Bus (PLM Side)

A.6.2.3 + 28 Volt Scanner Bus

A.6.2.3.1 Electrical Interface Description

A.6.2.3.1.1. PLM Side

The + 28 Volt Scanner Bus is a regulated bus, supplied to each instrument as required. This bus is classified as a well regulated bus, with minimum ripple and transients, and shall be used for such instrument equipment requiring power during launch and safe mode.
It is provided from the PLM by conversion and regulation of the 28 V unregulated power.
The Scanner Bus I/F circuit is shown in figure A.6.2.3.1-1.
The Scanner Bus power line is protected against short circuit by fuses and switched on the PLM side. Except for fault conditions or spacecraft emergency, the PLM switch shall not be used to switch the instrument rated power.
The Scanner Bus will automatically be switched on in case the spacecraft looses nominal attitude by failure.
The PCU provides redundant Scanner Power sources and fusing which will be selected on PLM side. Instruments will have a single Scanner Power connection utilising a single Scanner Power line.
Power line return is connected to the electrical ground on the PLM side.
The various Scanner Bus power outlets are protected by fuses to a maximal current as specified in the relevant ISAICD.

A.6.2.3.1.2. Instrument Side

See description in the relevant ISAICD.

A.6.2.3.2 Connectors

A.6.2.3.2.1. PLM Side

The connector type used for the Bus lines on PLM side is TBD.
The compatibility with the instrument side will be ensured by the PLM.

A.6.2.3.2.2. Instrument Side

The connector type used for the Scanner Bus lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for connection of several/all power buses.

A.6.2.3.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

T B D

Figure A.6.2.3.1-1: Circuit Diagram - Scanner Bus (PLM Side)

A.6.2.4 + 28 Volt Switched Telemetry Bus

A.6.2.4.1 Electrical Interface Description

A.6.2.4.1.1. PLM Side

The + 28 Volt Switched Telemetry Bus is a separate, switched bus supplied to each instrument. The primary function of this bus is to permit monitoring of the instrument temperature when the instrument is "OFF".
It is provided by the PLM by conversion and regulation of the 28 V unregulated power.
The Switched Telemetry Bus I/F circuit is shown in figure A.6.2.4.1-1.
All Switched Telemetry Buses are commonly protected against short circuit by a fuse and commonly switched on the PLM side. The fuse value is TBD.
The PCU provides redundant Switched Telemetry Bus sources and fusing which will be selected on PLM side. Instruments will have a single Switched Telemetry Bus connection utilising a single Switched Telemetry Bus line.
Power line return is connected to the electrical ground on the PLM side.

A.6.2.4.1.2. Instrument Side

See description in the relevant ISAICD.

A.6.2.4.2 Connectors

A.6.2.4.2.1. PLM Side

The connector type used for the + 28 V Switched Telemetry Bus lines on PLM side is TBD.
The compatibility with the instrument side will be ensured by the PLM.

A.6.2.4.2.2. Instrument Side

The connector type used for the + 28 V Switched Telemetry Bus lines on the instrument side are shown in the relevant ISAICD.
One common connector may be used for connection of several/all power buses.

A.6.2.4.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

TBD

Figure A.6.2.4.1-1: Circuit Diagram - Switched Telemetry Bus (PLM Side)

A.6.2.5 + 28 Volt Pulse Load Bus

A.6.2.5.1 Electrical Interface Description

A.6.2.5.1.1. PLM Side

The + 28 Volt Pulse Load Bus is a regulated bus, supplied to each instrument as required to power high current transient loads, such as stepper motors and heaters, which cannot meet the Main Bus current ripple and transient specifications. The Pulse Load Bus I/F circuit is shown in figure A.6.2.5.1-1.
It is provided by the PLM by conversion and regulation of the 28 Volt unregulated power.
The Pulse Load Bus line is protected against short circuit by fuses and switched on PLM side.
Except for fault conditions or spacecraft emergency, the PLM switch shall not be used to switch the instrument rated power.
The PCU provides redundant Pulse Load sources and fusing which will be selected on PLM side. Instruments will have a single Pulse Load connection utilising a single Pulse Load line.
Power line return is connected to the electrical ground on the PLM side.
The various Pulse Load Bus power outlets are protected by fuses to a maximal current as specified in the relevant ISAICD.

A.6.2.5.1.2. Instrument Side

See description in the relevant ISAICD

A.6.2.5.2 Connectors

A.6.2.5.2.1. PLM Side

The connector type used for the Pulse Load Bus lines on PLM side is TBD.
The compatibility with the instrument side will be ensured by the PLM.

A.6.2.5.2.2. Instrument Side

The connector type used for the Pulse Load Bus lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for connection of several/all power buses.

A.6.2.5.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

TBD

Figure A.6.2.5.1.-1: Circuit Diagram - Pulse Load Bus (PLM Side)

A.6.2.6 Active Thermal Control Heaters

The bus will be used by software controlled heaters inside the instruments, to keep the instrument within its nominal operation temperature range. It is switched inside the PCU under PMC control.

A.6.2.6.1 Electrical Interface Description

A.6.2.6.1.1. PLM Side

This bus is classified as a regulated + 28 Volt bus, with minimum ripple and transtents. The Active Thermal Control Heater bus I/F is shown in figure A.6.2.6.1-1.
It is provided by the PLM by conversion and regulation of the 28 Volt unregulated power.
The heater power is protected against short circuit by fuses and switched on PLM side under PMC control.
The PCU provides redundant Thermal Control Heater sources and fusing which will be selected on PLM side. Instruments will have a single Thermal Control Heater connection utilising a single Thermal Control Heater line.
Power line return is connected to the electrical ground on the PLM side.
The various Active Thermal Control Heater power outlets are protected by fuses to a maximal current as specified in table A.6.2.6.1-1.

Outlet [W]	Type of Fuse/Limiter	Maximum Specified Current
TBD	TBD	TBD

Table A.6.2.6.1-1: Active Thermal Control Heater Outlet Characteristics

A.6.2.6.1.2. Instrument Side

See description in the relevant ISAICD

A.6.2.6.2 Connectors

A.6.2.6.2.1. PLM Side

The connector type used for the Active Thermal Control lines on PLM side is TBD.
The compatibility with the instrument side will be ensured by the PLM.

A.6.2.6.2.2. Instrument Side

The connector type used for the Active Thermal Control Heater lines on the instrument side are shown in the relevant ISAICD.
One common connector may be used for connection of several/all power buses.

A.6.2.6.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

Figure A.6.2.6.1-1: Circuit Diagram - Active Thermal Control Heaters (PLM Side)

A.6.2.7 + 10 Volt Interface Bus

A.6.2.7.1 Electrical Interface Description

A.6.2.7.1.1. PLM Side

The + 10 Volt Interface Bus is a regulated bus at a level of 10 V ± 0.5 V and V_{ss} = 0 V at the source within the PLM. It shall be used only for powering the instruments control signal interfaces.
The + 10 V Interface Bus I/F circuit is shown in figure A.6.2.7.1-1.
It is provided by the PLM by conversion and regulation of the 28 V unregulated power.
The + 10 V Interface Bus is commonly protected against short circuit by one fuse device and is switched individually on PLM side.
The PLM provides redundant +10V sources, fuses, and switching.
The power line returns are connected to the electrical ground on the PLM side.

A.6.2.7.1.2. Instrument Side

See description in the relevant ISAICD.

A.6.2.7.2 Connectors

A.6.2.7.2.1. PLM Side

The connector type used for the Interface Bus lines on PLM side is TBD.
The compatibility with the instrument side will be ensured by the PLM.

A.6.2.7.2.2. Instrument Side

The connector type used for the Interface Bus lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for connection of several/all power buses, or together with the interface signal lines.

A.6.2.7.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

TBD

Figure A.6.2.7.1-1: Circuit Diagram - + 10 V Interface Bus (PLM side)

A.6.3 Command & Control Interface

Each C&C signal on the instrument side will be a single unbalanced bilevel signal, supplied from and received by standard COS/MOS (complementary-symmetry/metal-oxide-semiconductor) (CD4000 series) logic.
The interface circuits on instrument side shall be powered by a regulated power bus distributed by the satellite at a level of $V_{DD} = + 10.0 \pm 0.5$ volts and $V_{SS} = 0$ volts (reference).
The power and ground for the C&C interface shall be completely isolated from other power and grounds in the instrument.
For analog data acquisition by the satellite the instrument shall deliver the Analog TM Output and the Signal Ground (the latter one at least once on each analog signal connector (to be verified).
The NIU will provide all command and control interfaces to NOAA instruments and will be internally redundant.
The Remote Adaptation Unit (RAU) will adapt single ended interfaces to the PLM design and will be non-redundant.

A.6.3.1 Command & Control Interface Budgets

The individual instrument command & control interfaces and their required outlets are defined in the Instrument Specific Avionics ICD (ISAICD). The present baseline budget is shown in table A.6.3.1-1.

Instrument	Pulse Discrete CMD's	Level Discrete Cmd's	Analog TM	Thermistor Channels of +28 Switched TM Bus	Digital B TM
AVHRR-3	30 (60 ms)	0	22	0	15
AMSU A1	4 (60 ms)	10	21	6	12
AMSU A2	4 (60 ms)	9	11	4	10
HIRS-2	26 (60 ms)	0	14	2	14
DCS-2	24 (60 ms)	0	9	4	15
SEM	*7 (60 ms)*	*6*	*9*	*4*	*9*
SARR	*14 (60 ms)*	*18*	*13*	*9*	*4*
SARP-2	*22 (60 ms)*	*0*	*9*	*3*	*13*
Total	88/*(131)*	19/*(43)*	77/*(108)*	16/*(32)*	66/*(92)*

(..) = Numbers including SEM and S&R

Table A.6.3.1-1: Command & Housekeeping Data Budgets

A.6.3.2 Level Discrete Commands

A.6.3.2.1 Electrical Interface Description

A.6.3.2.1.1. PLM Side

This type of command presents an ON or TRUE condition to the instrument full time until another command is given to change the state to OFF or FALSE; i.e., it takes a specific command to change the state of this type of command. The command level is established by the state of one bit in a parallel output buffer in the NIU.
The instrument contractor may use a series of level discrete commands in a coded fashion if he so desires. For instance, four commands may be used to represent a 4-bit binary code. In this case, the instrument would receive four bilevel signal lines which would be decoded by the instrument into 16 states. To change states, a series of one to four sequential commands would be sent by the spacecraft. The bilevel signal lines would change state in the order commanded, with a time delay between successive line changes. Thus, the decoded output in the instrument would step through several states before achieving the final desired state. In order to eliminate this "state stepping", a fifth level or pulse discrete command could be used as a strobe. It is also possible to change multiple bits with a single command. Use of multiple-bit level commands, as described above, requires concurrence of the customer prior to implementation.
The standard PLM Level Discrete Command I/F circuit is shown in figure A 6.3.2.1.-1.

A.6.3.2.1.2. Instrument Side

The standard instrument side I/F circuit is shown in figure A.6.3.2.1-1. Deviations shall be shown in the relevant ISAICD.

Fig. A.6.3.2.1-1: Standard Level Discrete Command I/F Circuits

A.6.3.2.2 Connectors

A.6.3.2.2.1. PLM Side

The connector type and for the level Discrete Command lines on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM. One common connector may be used for various signals checks.

A.6.3.2.2.2. Instrument Side

The connector type used for the level discrete command lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for various signals/clocks.

A.6.3.2.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

A.6.3.3 Pulse Discrete Commands

A.6.3.3.1 Electrical Interface Description

A.6.3.3.1.1. PLM Side

In the pulse discrete command, an ON or TRUE condition is issued in the form of a pulse to the instrument over a single line. The pulse level is established by the state of one bit in a parallel output buffer in the PLM. Unlike the level discrete command, the pulse discrete command is automatically terminated (returned to the OFF state) by the PLM.
The pulse ON level duration is 60 ±5 ms. The pulse discrete command is normally used to change the state of a latching relay in the instrument.
The instrument shall not be damaged by a longer duration of the pulse than 65 ms.
The standard PLM Pulse Discrete Command I/F circuit is shown in figure A 6.3.3.1-1.

A.6.3.3.1.2. Instrument Side

The standard instrument side I/F circuit is shown in figure A.6.3.3.1-1 Deviations shall be shown in the relevant ISAICD.

A.6.3.3.2 Connectors

A.6.3.3.2.1. PLM Side

The connector type used for the Pulse Discrete Command lines on PLM side is TBD. The cmpatibility with the instrument side will be ensured by the PLM. One common connector may be used for various signals checks.

A.6.3.3.2.2. Instrument Side

The connector type used for the Pulse Discrete Command lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for various signals/clocks.

A.6.3.3.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

Fig. A.6.3.3.3-1: Standard Pulse Discrete Command I/F Circuits

A.6.3.4 Master Reference Clock

The instrument shall operate synchronously with reference to the clocks furnished from the PLM. The instruments will have a 1.248 MHz (0.994 for AVHRR) and a 1 Hz clock available from the PLM. One, or both, of these clocks shall be used to generate all timing needed for the instrument. All clock and sync signals are derived from the same frequency source with the same stability and drift characteristics.

Tabel A.6.3.4.-1 shows the timing diagram for the various clocks and sync signals

A.6.3.4.1 Electrical Interface Description

A.6.3.4.1.1. PLM Side

1.248 Megahertz Clock

(1) Signal type: constant frequency symmetrical trapezoidal wave
(2) Frequency: 1.248 MHz
(3) Short term stability:
±5 parts in 10^9 per minute
(4) Frequency drift:
±3 parts in 10^8 per week; ±2 parts in 10^6 per year.
(5) The signal symmetry will be better than 82 % for a standard CMOS receiver interface
(6) The 1.248 MHz line shall utilize the high speed interface with the spacecraft as shown in figure A.6.3.4.1.-1.
(7) The 1.248 MHz clock is distributed from the NOAA Interface Unit (NIU).

0.9984 Megahertz Clock (to be used only by AVHRR instrument)

(1) Signal type: constant frequency symmetrical trapezoidal wave
(2) Frequency : 0.9984 MHz
(3) Stability and symmetry:
The stability is the same as for the 1.248 MHz clock.
The symmetry for a standard TTL receiver shall be better than 67 %.
(4) Interface: Special interface defined in AVHRR ISAICD and shown in figure A.6.3.4.1.-2.
(5) Source: The 0.9984 MHz clock is distributed to the MIRP and TIP from Cross Strap Unit. The clock is sent to the AVHRR from the NIU.

A.6.3.4.1.2. Instrument Side

The standard instrument side I/F circuit is shown in figure A.6.3.4.1-1. Deviations shall be shown in the relevant ISAICD. For AVHRR it is shown in fig. A 6.3.4.1-2.

A.6.3.4.2 Connectors

A.6.3.4.2.1. PLM Side

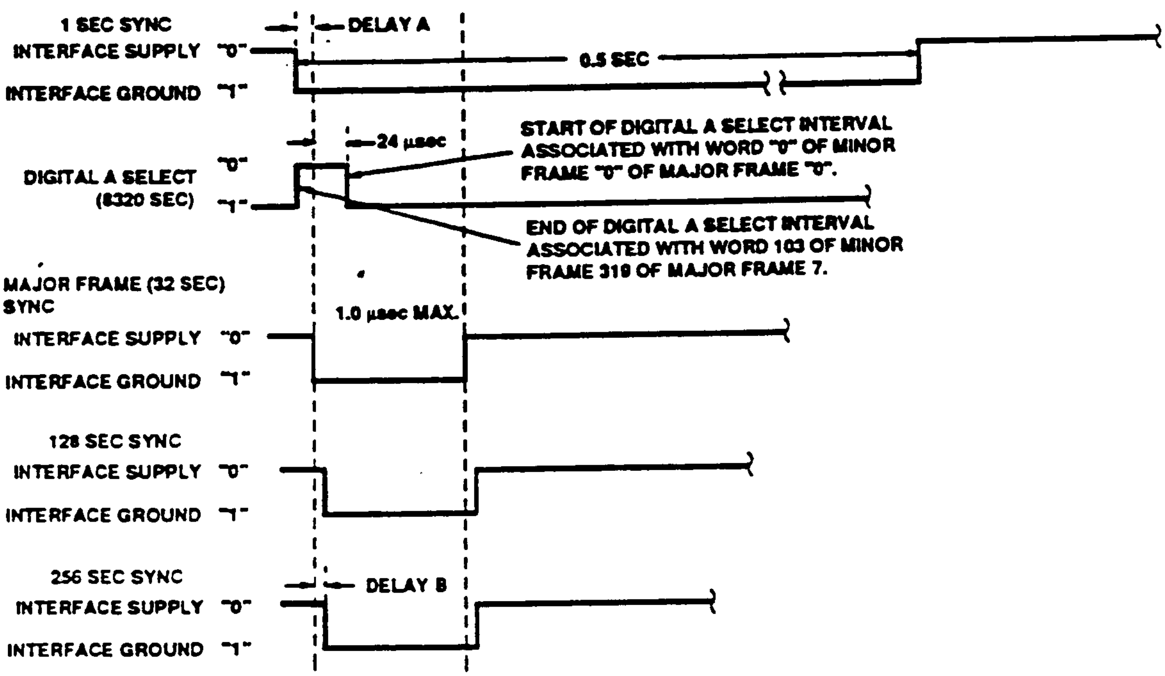
The connector type used for the Master Reference Clock lines on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM. One common connector may be used for various signals checks.

A.6.3.4.2.2. Instrument Side

The connector type used for the Master Reference Clock lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for various signals/clocks.

A.6.3.4.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.



- DELAY A = 4.0 µsec MAX, DELAY B = 1.0 µsec MAX AND THE SUM OF DELAY A + DELAY B CANNOT EXCEED 4.5 µsec
- LOGIC "1" (ACTIVE LEVEL) - GROUND
- STANDARD FAST OR SLOW INTERFACES SHALL BE USED FOR TRANSFER OF SYNC SIGNALS
- GROUND REFERENCED TO INTERFACE GROUND

Fig. A.6.3.4.-1: Timing Diagram-Sync Signals (with 8.32 kbps digital A Data Rate)

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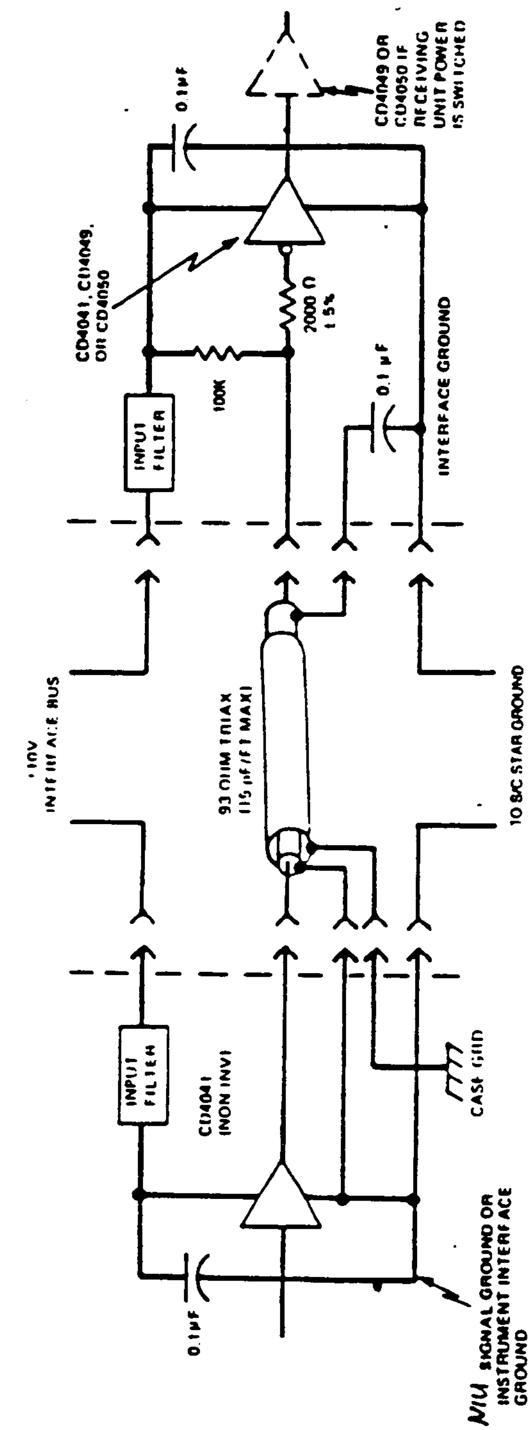

Fig. A.6.3.4.1-1: Master Reference Clock I/F (TBC)

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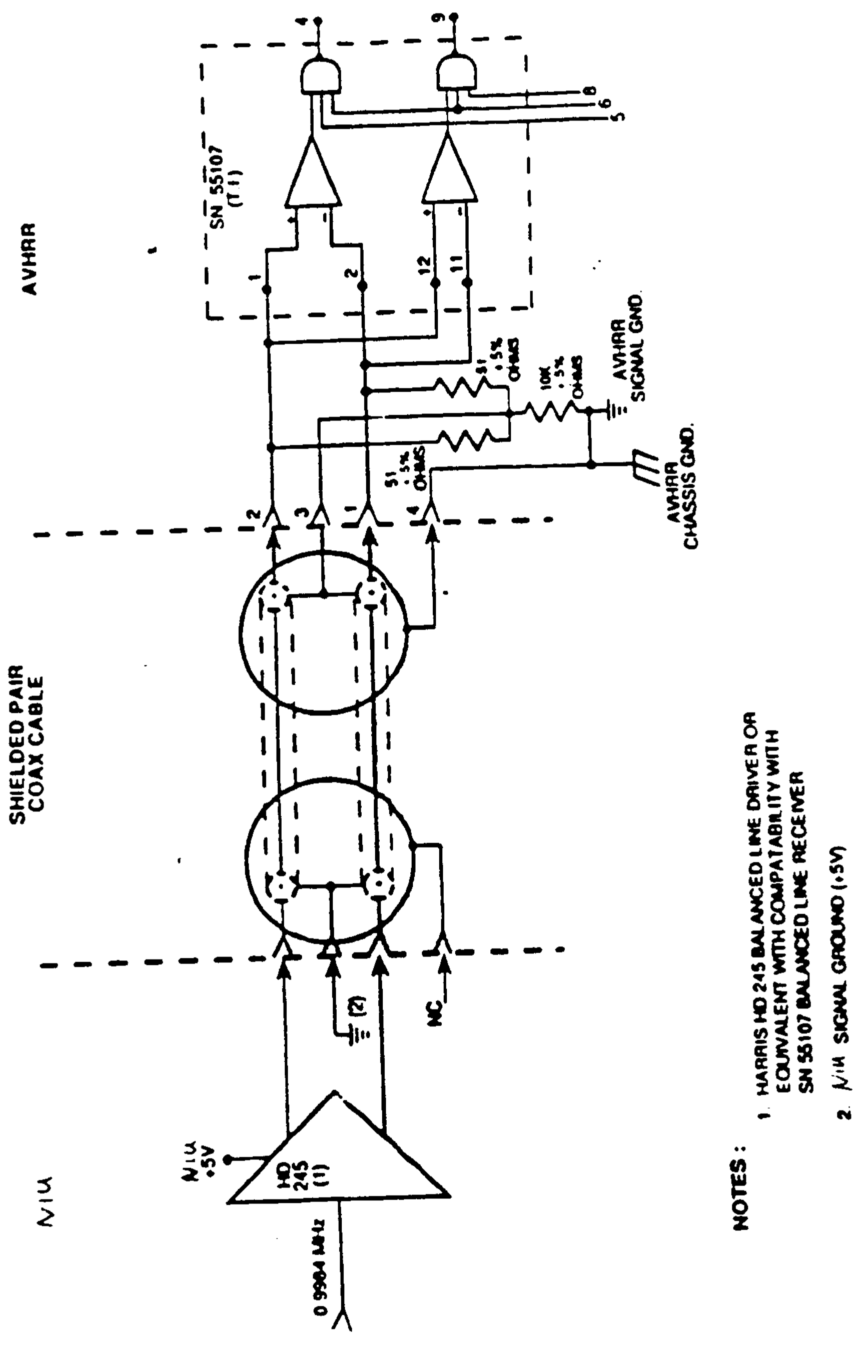


Fig. A.6.3.4.1-2: AVHRKR 0,9984 MHz Master Reference Clock I/F (TBC)

A.6.3.5 1-Hz Clock

A.6.3.5.1 Electrical I/F Description

A.6.3.5.1.1. PLM Side

The 1 Hz clock signal is derived from the same Master Reference source as the 1.248 MHz Clock signal and as such will have the same stability and drift characteristics. It is recommended that one of the hysteresis circuits be used to interface this clock with the spacecraft.
The wave shape shall be of constant frequency and shall be a symmetrical trapezoidal wave. Symmetry shall be greater than 90 %. (See figure A.6.3.4-1.) Since its symmetry may vary with temperature, it is recommended that instruments use only one edge of this signal. The 1 Hz clock is distributed from the NIU.
The standard I/F circuit diagram is shown in fig. A. 6.3.5.1-1.

A.6.3.5.1.2. Instrument Side

The standard instrument side I/F circuit is shown in figure A.6.3.5.1-1. Deviations shall be shown in the relevant ISAICD.

A.6.3.5.2 Connectors

A.6.3.5.2.1. PLM Side

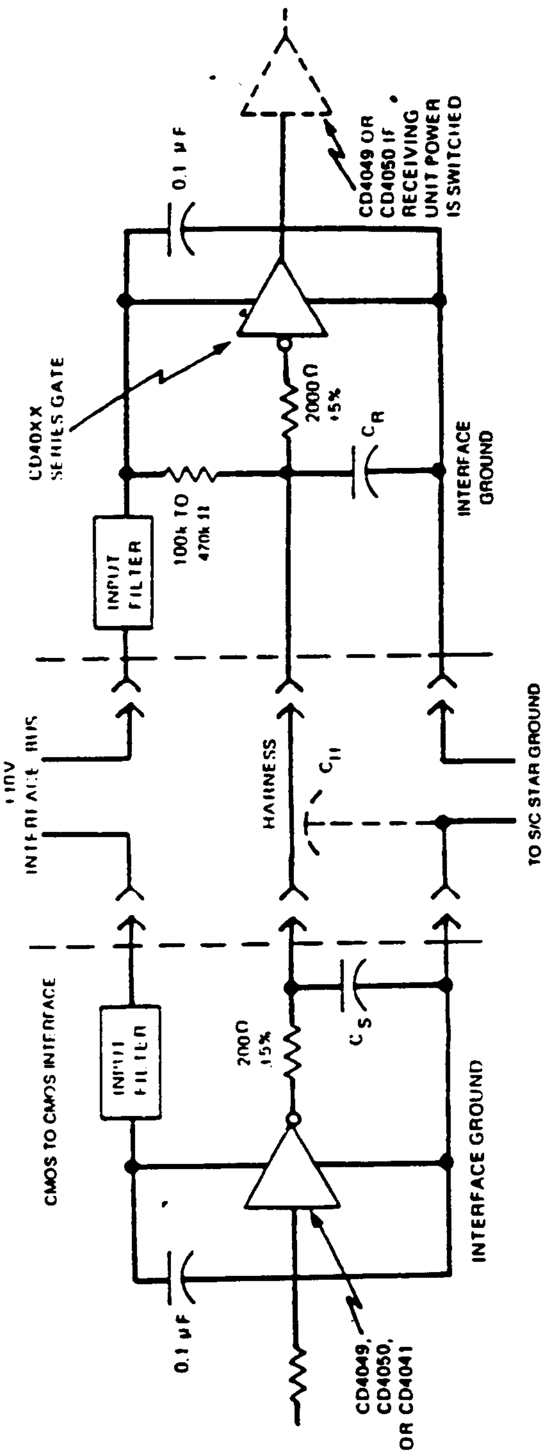
The connector type used for the 1 Hz Clock lines on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM. One common connector may be used for various signals checks.

A.6.3.5.2.2. Instrument Side

The connector type used for the 1 Hz clock lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for various signals/clocks.

A.6.3.5.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.



CMOS TO CMOS INTERFACE CHARACTERISTICS

CKT TYPE	C_S (pF)	C_H (pF)	TR/TF MIN (µs)	TR/TF MAX (µs)	TD MAX (µs)
SLOW	5600 ±10%	5600 ±10%	5.0	12.0	8.0
FAST	270 ±5%	470 ±5%	0.3	2.0	1.0

NOTE MAX VALUES INCLUDE 500 pF CABLE CAPACITY (C_H)

Fig. A.6.3.5.1-1 1 Hz Clock I/F (TBC)

A.6.3.6 Major Frame Sync Pulse

The instrument will be furnished with a major frame synchronization pulse to synchronize the complete system including the scan. The major frame pulse is 240,4 in length. The major frame pulse occurs once every thrity-two seconds.

A.6.3.6.1 Electrical I/F Description

A.6.3.6.1.1. PLM Side

The PLM side I/F circuit diagram is shown in fig. A 6.3.6.1-1.

A.6.3.6.1.2. Instrument Side

The receiver interface for the calibration pulses shall be the standard Slow or Fast interfaces. as shown in fig. A.6.3.6.1-1. Deviations shall be shown in the relevant ISAICD.

A.6.3.6.2 Connectors

A.6.3.6.2.1. PLM Side

The connector type used for the Major Frame Sync Pulse lines on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM. One common connector may be used for various signals checks.

A.6.3.6.2.2. Instrument Side

The connector type used for the Major Frame Sync Pulse lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for various signals/clocks.

A.6.3.6.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

A.6.3.7 128/256 Seconds Calibration Pulse

The instrument will be furnished a 240.4 µsecec pulse occurring every 128 seconds or every 256 seconds. The logic "0" to logic "1" transition of this pulse is within 1 µsec of the negative going edge of the major frame synchronization pulse. This calibration pulse shall initiate a scan line count for instruments with stepper mirrors.
It will be derived from the same source as the 1.248 MHz Master Reference Clock and as such have the same stability and drift characteristics. The timing is shown in figure A.6.3.3-1.

A.6.3.7.1 Electrical I/F Description

A.6.3.7.1.1. PLM Side

The PLM side I/F circuit diagram is shown in fig. A.6.3.7.1.-1.

A.6.3.7.1.2. Instrument Side

The receiver interface for the major frame synchronization pulse shall be standard Slow or Fast interfaces as shown in figure A.6.3.7.1-1. Deviations shall be shown in the relevant ISAICD.

A.6.3.7.2 Connectors

A.6.3.7.2.1. PLM Side

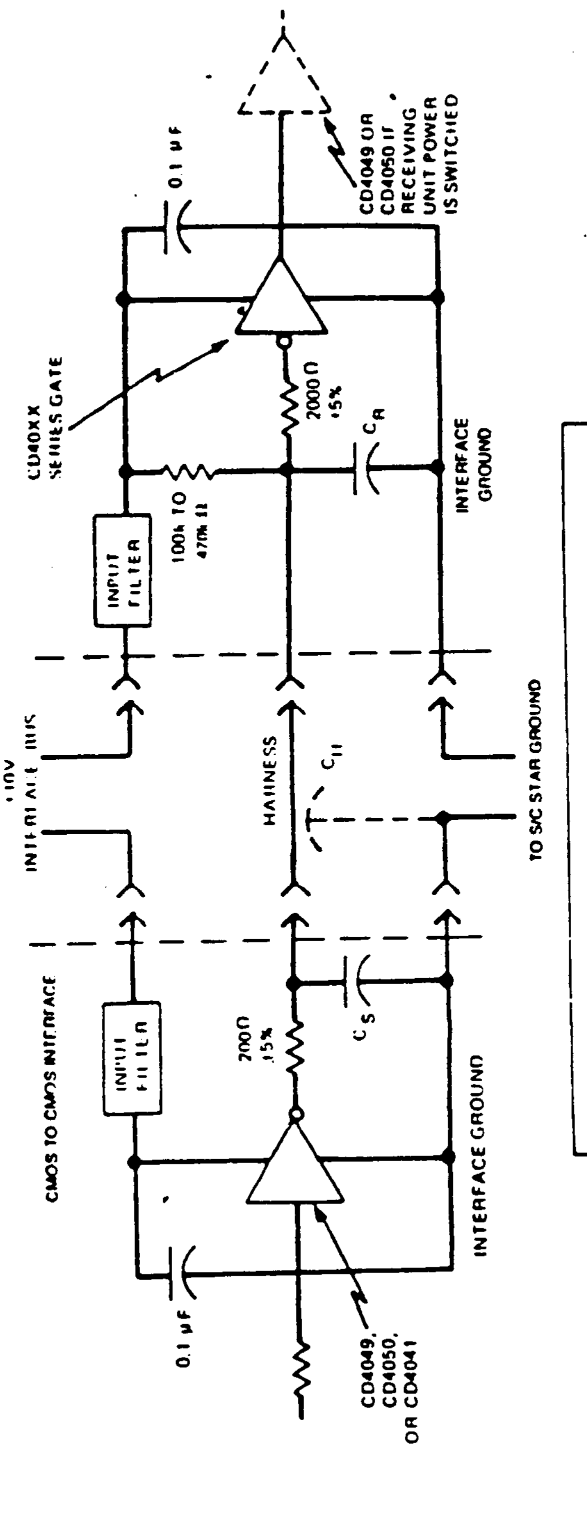
The connector type and for the Calibration Pulse lines on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM. One common connector may be used for various signals checks.

A.6.3.7.2.2. Instrument Side

The connector type used for the Calibration Pulse lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for various signals/clocks.

A.6.3.7.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.



CMOS TO CMOS INTERFACE CHARACTERISTICS

CKT TYPE	C_S (pF)	C_H (pF)	TR/TF MIN (µs)	TR/TF MAX (µs)	TD MAX (µs)
SLOW	5600 ±10%	5600 ±10%	5.0	12.0	8.0
FAST	270 ±5%	470 ±5%	0.3	2.0	1.0

NOTE: MAX VALUES INCLUDE 500 pF CABLE CAPACITY (C_H)

Fig. A.6.3.7.1-1: Calibration Pulse I/F Circuit Diagram (TBC)

A.6.3.8 8 sec Sync Pulse

The 8 sec sync pube is used to synchronise AMSU-A1, f A2 and MHs. It will be derived from the same source as the 1.248 MHz Master Reference Clock and as such have the same stability and drift characteristics. The timing is shown in figure A.6.3.3-1

A.6.3.8.1 Electrical I/F Description

A.6.3.8.1.1. PLM Side

The PLM side I/F circuit diagram is shown in fig. A.6.3.8.1-1.

A.6.3.8.1.2. Instrument Side

The standard instrument side I/F circuit is shown in figure A.6.3.8.1.1-1. Deviations shall be shown in the relevant ISAICD.

A.6.3.8.2 Connectors

A.6.3.8.2.1. PLM Side

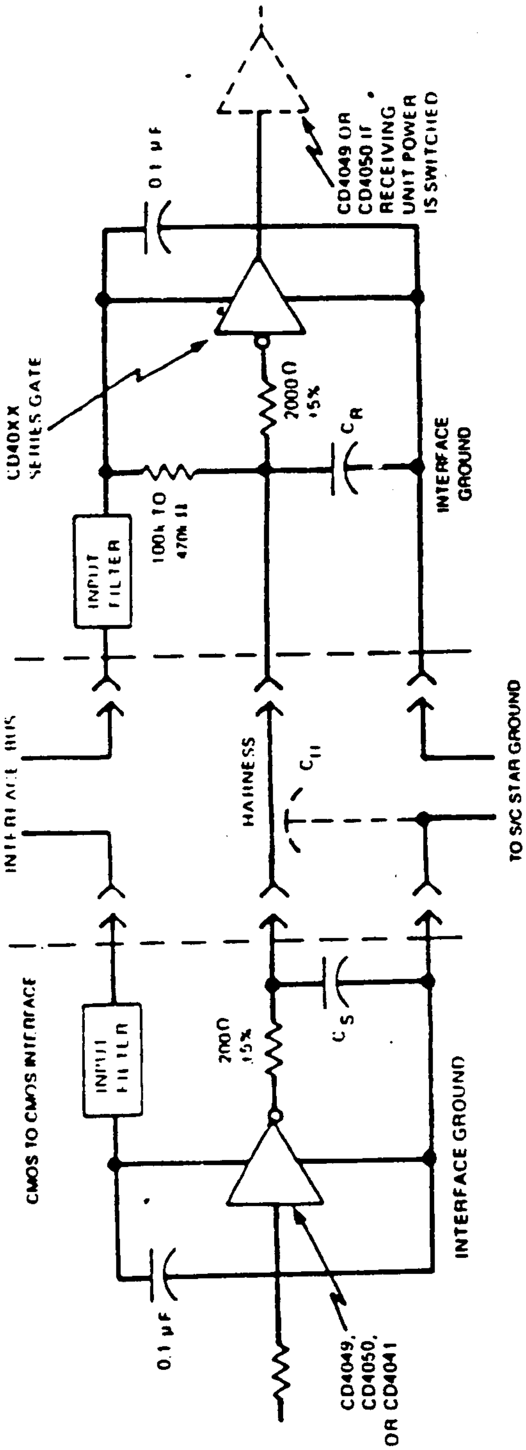
The connector type used for the 8 sec Sync Pulse lines on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM. One common connector may be used for various signals checks.

A.6.3.8.2.2. Instrument Side

The connector type used for the 8 sec Sync Pulse lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for various signals/clocks.

A.6.3.8.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.



CMOS TO CMOS INTERFACE CHARACTERISTICS

CKT TYPE	C_S (pF)	C_H (pF)	TR/TF MIN (µs)	TR/TF MAX (µs)	TD MAX (µs)
SLOW	5600 ±10%	5600 ±10%	5.0	12.0	8.0
FAST	270 ±5%	470 ±5%	0.3	2.0	1.0

NOTE MAX VALUES INCLUDE 500 pF CABLE CAPACITY (C_H)

Fig. A.6.3.8.1-1: 8 sec-Sync Pulse I/F Circuit Diagram (TBC)

A.6.3.9 Analog Housekeeping

Analog housekeeping telemetry points monitor key temperatures and voltages from NOAA instruments. The resolutions of the analog telemetry, as received on the ground, is one (1) part in 256 (20 mV).

A.6.3.9.1 Electrical Interfaces

A.6.3.9.1.1. PLM Side

The PLM side will be compliant to the requirements shown in table A 6.3.9.1-1. The PLM side interface circuit diagram is shown in fig. A 6.3.9.1-1.

A.6.3.9.1.2. Instrument Side

Analog telemetry circuits shall be powered from the secondary side of the instrument DC/DC converter. Analog telemetry signals supplied from any instrument shall comply with the requirements listed in table A 6.3.9.1-1

Parameter	Value
Signal Type:	analog
Signal Level:	0 to +5.12 V referenced to DC signal ground
Frequency Response:	DC to 200 Hz
Output Configuration:	Single-ended, DC coupled, referenced to signal ground (signal ground outlet at same connector for differential acquisition)
Output Impedance:	2.0 k < R < 15.0
For Ground Return, Source/Sink Current, Overrange Limits a.s.o.	Refer to ID1

Table A.6.3.9.1-1: Analog Telemetry Characteristics

Failure Mode Protection. Each Analog Housekeeping telemetry point shall be so configured that the continuous application of voltage in the range of -10 V to +10 V through zero source impedance (in the NIU) shall not cause the instrument, within which the telemetry point is located, to malfunction in any way. However, the telemetry signal need not be valid during this failure mode.
The standard instrument side I/F circuit is shown in figure A.6.3.9.1-1. Deviations shall be shown in the relevant ISAICD.

A.6.3.9.2 Connectors

A.6.3.9.2.1. PLM Side

The connector type and for the Analog Housekeeping lines on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM. One common connector may be used for various signals.

A.6.3.9.2.2. Instrument Side

The connector type used for the Analog Housekeeping lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for various signals.

A.6.3.9.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

TBD

Fig. A 6.3.9.1-1: Analog telemetry I/F Circuit Diagram

A.6.3.10 Digital Housekeeping (Digital B)

Digital housekeeping telemetry (also referred to as discrete, bilevel, or Digital B) consists of single bit digital status "flags" for relays and operation mode monitors. All discrete bilevel monitors will be sampled at least once every 3.2 seconds.

A.6.3.10.1 Electrical Interface Description

A.6.3.10.1.1. PLM Side

The PLM side will be compliant to the requirements shown in table A 6.3.10.1-1. The PLM side interface circuit diagram is shown in fig. A 6.3.10.1-1.

A.6.3.10.1.2. Instrument Side

Digital Housekeeping signals shall comply with the requirements listed in table A.6.3.10.1-1.

The standard instrument side I/F circuit is shown in figure A.6.3.10.1-1. Deviations shall be shown in the relevant ISAICD.

Parameter	Value
Signal Level	
Data '1' (Low Voltage STate) Data '0' (High Voltage State)	-0.1 to +0.5 V +3.5 to +5.7 V
Frequency Response:	DC to 200 Hz
Output Configuration:	Single-ended, DC coupled, referenced to 'Star' ground
Output Impedance:	2.0 k1/2 < R < 15.0 k1/2
For Ground Return, Source/Sink Curent, Overrange Limits a.s.o.:	Refer to ID1

Table A.6.3.10.1-1: Digital B Telemetry Characteristics

A.6.3.10.2 Connectors

A.6.3.10.2.1. PLM Side

The connector type used for the Digital B telemetry lines on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM. One common connector may be used for various signals checks.

A.6.3.10.2.2. Instrument Side

The connector type used for the Digital B telemetry lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for various signals/clocks.

A.6.3.10.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

TBD

Fig. A.6.3.10.1-1:Digital B Telemetry I/F Circuit Diagram

A.6.3.11 Switched Telemetry Bus Thermistor Interface

A.6.3.11.1 Electrical Interface Description

The Switched Telemetry Bus Thermistors shall provide essential instrument temperatures when the instrument is off.
Temperature monitoring circuits other than simple resistor/thermistor networks are not allowed.

A.6.3.11.1.1. PLM Side

The PLM side is TBD. The interface circuits diagram is shown in fig. A 6.3.11.1-1.

A.6.3.11.1.2. Instrument Side

The I/F requirements are TBD!!.
The standard instrument side I/F circuit is shown in figure A.6.3.11.1-1 Deviations shall be shown in the relevant ISAICD.

A.6.3.11.2 Connectors

A.6.3.11.2.1. PLM Side

The connector type used for the Switched telemetry bus thermistor lines on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM. One common connector may be used for various signals.

A.6.3.11.2.2. Instrument Side

The connector type used for the switched telemetry bus thermistor lines on the instrument side are shown in the relevant ISAICD. One common connector may be used for various signals.

A.6.3.11.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

TBD

Fig. A.6.3.11.1-1:Switched Telemetry Bus Thermistor I/F Circuit Diagram

A.6.4 Measurement Data Interface

Instrument Measurement Data (Digital A data) outputs will be received by the NIU on single output lines. These signals will be cross-strapped internal to the NIU such that a single Digital A input gate failure will not impact other Digital A channels. The standard fast interface circuits will be used for transfer of all Digital A signals (see fig. 6.4.1.1-1).
The Digital A interface consists of a data line (D1) from the instrument to the NIU, and a clock (C_1) and data select interval (A_1) from the NIU to the instrument. The phasing of these signals is described in the Interface Timing Diagram shown in Fig. A 6.4-1.
The clock signal (C_1) is continuously available to all Digital A users at 8.32 kbps rate. The select interval (A_1) is enabled only when a particular Digital A data source is to be interrogared. (When not selected, the Digital A data output interface (D_1) shall be in the logic "0" inactive state). When interrogated, 8 bits of data are shifted serially to the NIU for readout in the NIU output data stream.
For AMSU-A1 and -A2 the C1 clock signal will be 16.64 kHz with 12 pulse width via the same I/F type (standard fast interface).

MSB Timing
The first (most significant) bit of information to be transferred shall be established; i.e., reached 90 % of value, a maximum of 48 before the logic "0" to logic "1" transition of the "C_1" shift pulse signal occurs or, as a minimum, coincident with the logic "0" to logic "1" transition of the first "C_1" shift pulse signal. The NIU clocks (accepts) the data on the logic "1" to logic "0" transition of the "C_1" shift pulse.

Second Through Eight Bit Timing
The second through eight most significant bits shall be established by the logic "0" to logic "1" transition of the remaining seven "C_1" shift pulses. The time delay, from the point at which the above transition is 70 % complete to the point at which the transition, if any, of the new data bit being shifted into the instrument output buffer is 90 % complete, shall be less than 6.0 µsec. These bits shall remain at their levels until the logic "0" to logic "1" transition of the next "C_1" shift pulse. The NIU clocks the data on the logic "1" to logic "0" transition of the "C_1" shift pulse.

Continuous Sampling of the Same Data Source
If "n" 8-bit samples of Digital A input occur consecutively in a sampling sequence, the "A_1" pulse generated by the NIU shall be "n" sample intervals wide with a 24 sec transition pulse occurring between each 8-bit sampling interval are shown in fig. A.6.4-1.

A.6.4.1 Electrical Interface Description

A.6.4.1.1 PLM Side

The PLM side interface circuits diagram is shown in fig. A.6.4.1-1., for the AMSU-A1 and -A2 in figure A6.4.1-2.

A.6.4.1.2 Instrument Side

The standard instrument side I/F circuit is shown in figure A.6.4.1-1 and -2. Deviations shall be shown in the relevant ISAICD.

A.6.4.2 Connectors

A.6.4.2.1 PLM Side

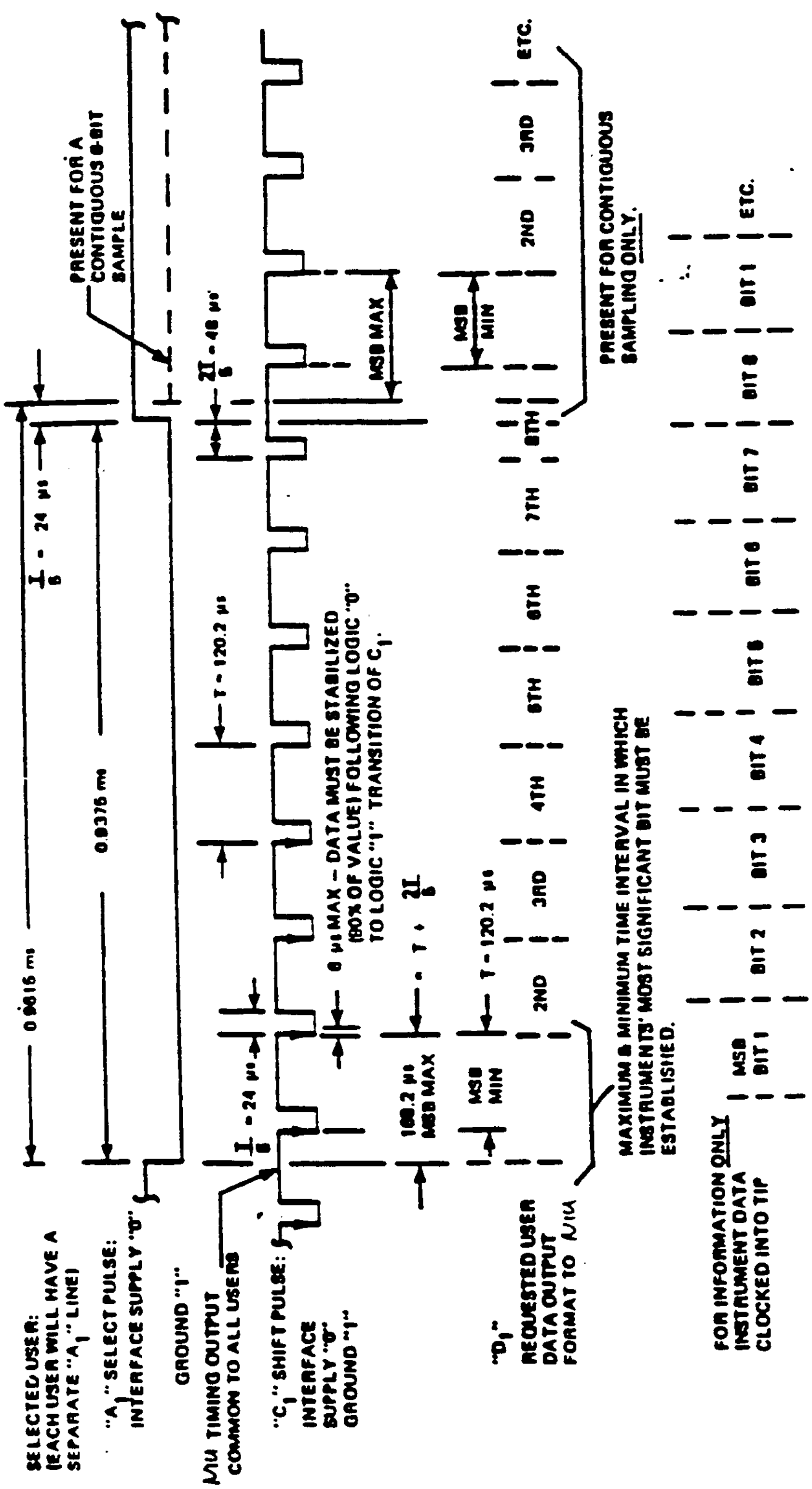
The connector type used for the 1 Hz Clock lines on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM. One common connector shall be used for various signals checks.

A.6.4.2.2 Instrument Side

The connector type used for the Measurement Data (Digital A) lines on the instrument side are shown in the relevant ISAICD. One common connector shall be used for various signals/clocks.

A.6.4.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.



NOTES:
- LOGIC "1" (ACTIVE LEVEL) = GROUND
- INSTRUMENT DIGITAL "A" DATA OUTPUT INTERFACE TO BE ACTIVE ONLY DURING "A$_i$" SELECT INTERVAL.
- TIROS-N STANDARD FAST INTERFACE WILL BE USED FOR TRANSFER OF ALL DATA AND CONTROL SIGNALS.
- GROUND REFERENCED TO INTERFACE GROUND.
- THE "A$_i$" PULSE ASSOCIATED WITH WORD 0 OCCURS AT THE START OF THE MINOR FRAME.

Fig. A.6.4-1: Measurement Data I/F Timing Diagram

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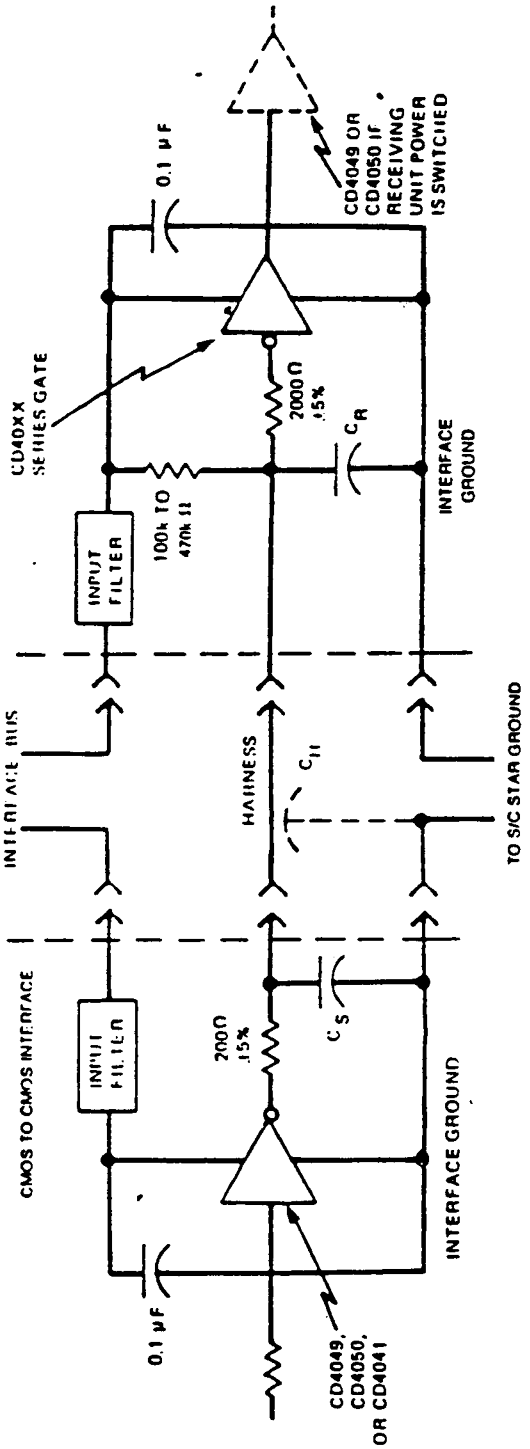
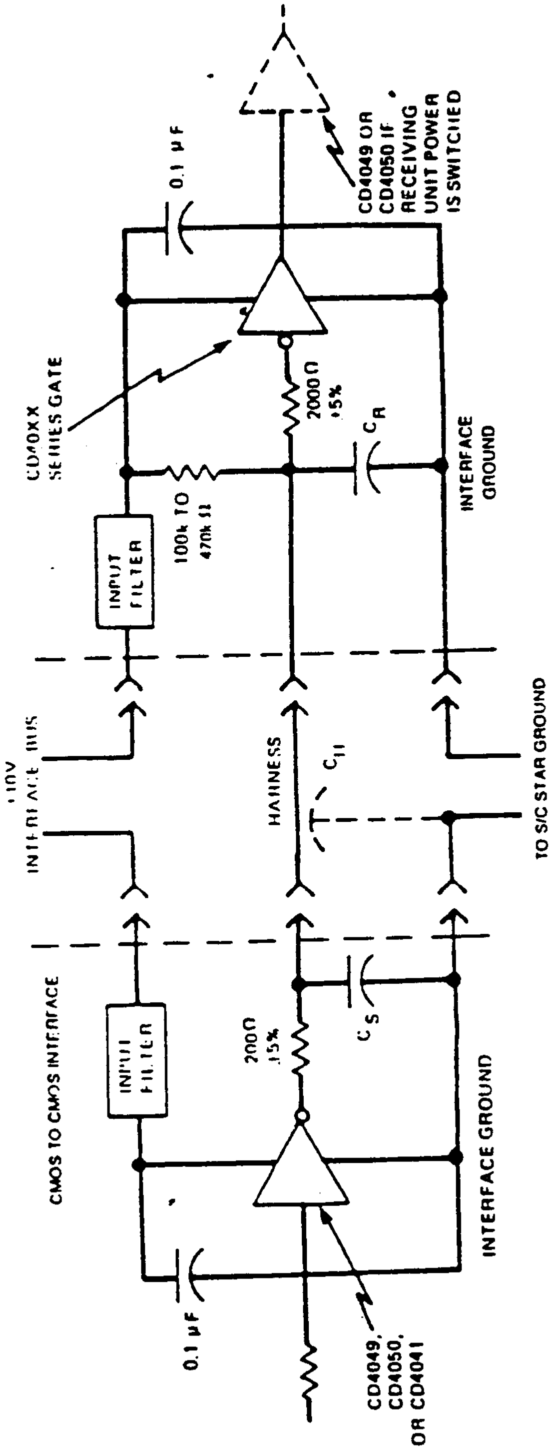


CMOS TO CMOS INTERFACE CHARACTERISTICS					
CKT TYPE	C_S (nF)	C_H (nF)	TR/TF MIN (µs)	TR/TF MAX (µs)	TD MAX (µs)
SLOW	5000 ±10%	5000 ±10%	5.0	12.0	8.0
FAST	220 ±5%	470 ±5%	0.3	2.0	1.0

NOTE MAX VALUES INCLUDE 500 nF CABLE CAPACITY (C_H)

Fig. A.6.4.1.-1: Measurement Data Interface for 8320 bps Data Rate



CMOS TO CMOS INTERFACE CHARACTERISTICS

CKT TYPE	C_S (pF)	C_H (pF)	TR/TF MIN (µs)	TR/TF MAX (µs)	TD MAX (µs)
SLOW	5600 ±10%	5600 ±10%	5.0	12.0	8.0
FAST	220 ±5%	470 ±5%	0.3	2.0	1.0

NOTE: MAX VALUES INCLUDE 500 pF CABLE CAPACITY (C_H)

Fig. A6.4.1-2 Measurement Data Interface for 16 640 bps Data Rate

A.6.5 Test Interfaces

The PLM will normally not provide access to test points on the instrument. In cases where access is required to a test point connector when the instrument is mounted on the PLM, this information shall be indicated in the ISAICD.

A.6.5.1 Safety Test Heaters

A.6.5.1.1 Electrical Interface Description

A.6.5.1.1.1. PLM Side

Safety Test Heaters will only be served from PLM side on special request in the ISAICD.

A.6.5.1.1.2. Instrument Side

See description in the relevant ISAICD

A.6.5.1.1.3. External Side

The Safety Heater bus is a regulated + 28 Volt (TBC) spacecraft external bus. It might be utilized for special test heaters during spacecraft-level thermal/vacuum testing if considered necessary.

A.6.5.1.2 Connectors

A.6.5.1.2.1. PLM Side

The connector type used on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM

A.6.5.1.2.2. Instrument Side

See relevant ISAICD

A.6.5.1.3 Pin Allocation

See relevant ISAICD

A.6.5.2 Test Points

A.6.5.2.1 Electrical Interface Description

A.6.5.2.1.1. PLM Side

Test points shall be concentrated in physically separate connectors (female). From PLM side test connectors are kept accessible as far as possible. Details are regulated in the ISAICD.

A.6.5.2.1.2. Instrument Side

Short Circuit Protection
Test points shall be internally buffered to prevent damage to the instrument in the event that a test point is shorted to ground.

Test Point Signal Characteristics
(a) Signal Type - Same as signal being monitored:
(b) Signal Amplitude - Representative of signal being monitored.
(c) Output Impedance - 10K Ohms, max.
(d) Output Configuration - Single ended, one pin per test point output.

A.6.5.2.2 Connectors

A.6.5.2.2.1. PLM Side

The connector type used for the test point lines on PLM side is TBD. The compatibility with the instrument side will be ensured by the PLM. One common connector shall be used for all test point signals.

A.6.5.2.2.2. Instrument Side

The connector type used for the test point lines on the instrument side are shown in the relevant ISAICD. One common female connector shall be used for all test point signals.

A.6.5.2.3 Pin Allocation

The pin allocation on instrument side is shown in the relevant ISAICD.

A.7 EMC

A.7.1 EMC Design

A.7.1.1 Electrical Bonding

A.7.1.1.1 General

Bonding is the method by which adjacent conductive elements are electrically connected in order to minimize any potential differences. Joint faces shall be flat and clean before assembly. The bond shall be resistant against corrosion. The only permitted surface finishes to joint faces are:

- clean metal (except aluminium and magnesium)
- gold plate on the base material
- alodine 1200 or similar

In general the bonding resistance shall be less than 10 mΩ (if not otherwise specified).

A.7.1.1.2 Secondary Power Starpoint

At the designated location where the secondary power return is referenced to structure (within the box) the DC resistance between the secondary power return line and the structure shall be less than 2.5 mΩ.

A.7.1.1.3 Equipment Cases

If instrument grounding will be performed via the box feets directly to the METOP ground reference the contact area of the underside of each feet shall not be less than 1 cm^2. The bonding resistance shall be less than 10 mΩ.
If the grounding of the instrument cannot be performed via the box feeds (i.e. for instruments which require thermal isolation from the METOP ground reference resp. instruments which are mounted on CFRP) the electrical contact shall be accomplished utilizing bonding straps of rectangular crossection. The width : length ratio of this strap shall be at least 1 : 5. The strap thickness has to be selected to bear the maximum expected fault current. The total DC resistance of such bond connection between case and METOP ground reference shall be ≤ 10 mΩ, measured with both directions of voltage polarity.
Remark: Under consideration of the early project phase (configuration not yet frozen) each particular instrument should provide a bonding stud for grounding purposes.

The bonding stud should have the following dimensions:

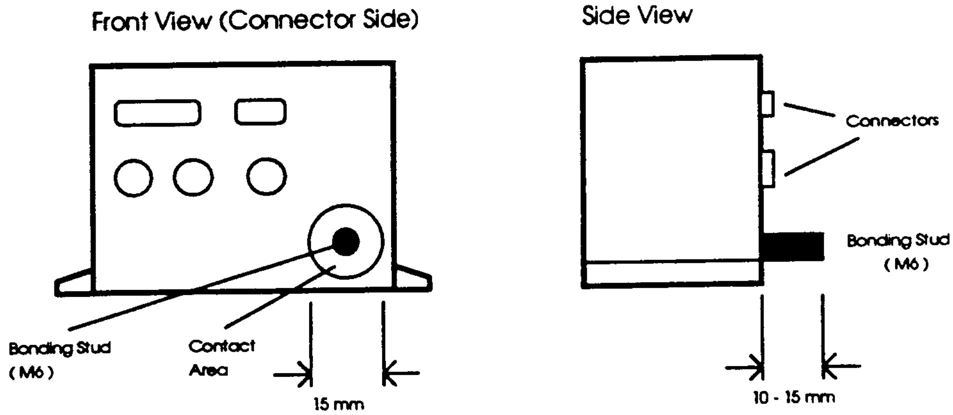


Fig. A.7.1.1.3- 1 Dimensioning of the Bonding Stud

A.7.1.1.4 Connectors

The connector shell of the box mounted connector part (= receptacle) shall be bonded to the equipment case with a DC resistance of < 10 mΩ. The connector backshell as part of the cable shield shall exhibit a DC resistance of ≤ 10 mΩ to the equipment case when connected. The resistance between any shield ground pin and the equipment case shall be less than 10 mΩ. This connection has to be as short as possible in order to minimize its effectiveness to act as an antenna which receives or transmits shield currents.

A.7.1.1.5 Cable and Harness Shields

The overall harness shields will be bonded to the unit cases resp. connector brackets via the connector backshell / connector receptacle or by a dedicated bond strap. The preferred method is to use the connector backshell.
The overall harness shield shall also be grounded to structure at intermediate points, where the harness is mechanically fixed to the structure. The DC resistance across the bond between any shield and the shield ground point (at the connector, at the PCB or at intermediate points) shall be ≤ 10 mΩ, measured for both directions of voltage polarity.

A.7.1.1.6 Structural Parts

All mechanical equipment/ components (including CFRP parts) which do not perform any electrical function shall be grounded to the structure by a bonding resistance of < 1 kΩ. between adjacent parts and to the local metallic ground reference.

Regarding Secondary Surface Mirrors (SSMs) the DC resistance between the metallized surface and another conductive point on the CFRP (Carbon Fibre Reinforced Plastics) skin shall be < 1 kΩ. The SSM tape width shall not be greater than 25 mm.

All Optical Solar Reflectors (OSRs) shall be attached to the substrate with conductive adhesive. Individual OSRs shall have a surface area of less than 17 cm².

Each electrically conductive layer of MLI shall be grounded to the METOP ground reference to avoid electro-static charging. The bonding resistance between blanket bond joint (rivet) and structure shall be shall be < 1 Ω. The bonding resistance between blanket and blanket bond joint (rivet) shall be < 1 Ω (to be measured during manufacturing of the MLI).

A.7.1.2 Grounding

The grounding concept of METOP is the Distributed Single Point Grounding concept (DSPG):
- Primary power must be isolated from both instrument case (structure) and internally generated secondary power (prevention of ground loops).
- Internally generated power must be low ohmic bonded to the instrument case.
- The instrument case must be low ohmic bonded to the structure (via box feets or dedicated bond strap).
- Single-ended I/F's not permitted. => Use single ended drivers & differential receivers (or transformers, opto-couplers).

Since the NOAA KLM instruments are originally not compatible with the DSPG but with the SPG (Single Point Grounding) system as used for the TIROS spacecraft, specific measures must be applied for fitting them into the PLM grounding architecture. These measures are:
1. The application of a NOAA instrument Interface Unit (NIU) and a Remote Adapter Unit (RAU). Both units provide for the required single ended interfaces to the specific KLM instruments. Since single ended interfaces will be very sensitive for common mode noise coupling, therefore the cable length between such interfaces have to be minimized. In such cases where the NIU cannot be located in the direct vicinity of the particular KLM instrument (i.e. for AMSU A2 and DCS), an additional RAU is required to be located close to the KLM instrument. The RAU provides for single ended interfaces to the KLM instruments and differential interfaces to the NIU.
2. The regulated 28 VDC as generated within the PCU will be referenced to structure as close to its source (=secondary side of the PCU converter) as possible via the lowest possible impedance path. This ground reference is the central ground reference (i.e. starpoint) for the KLM instruments.
3. In order to avoid groundloops between the KLM instrument, the NIU/ RAU and the PCU, the power used for the single ended interfaces within the NIU resp. RAU must be isolated from

the structure (floating). They are referenced to the structure within the PCU. The only permitted structure reference for power used by the KLM instruments is within the PCU.

Fig. A.7.1.2-1 shows the grounding and isolation scheme of the KLM instruments.

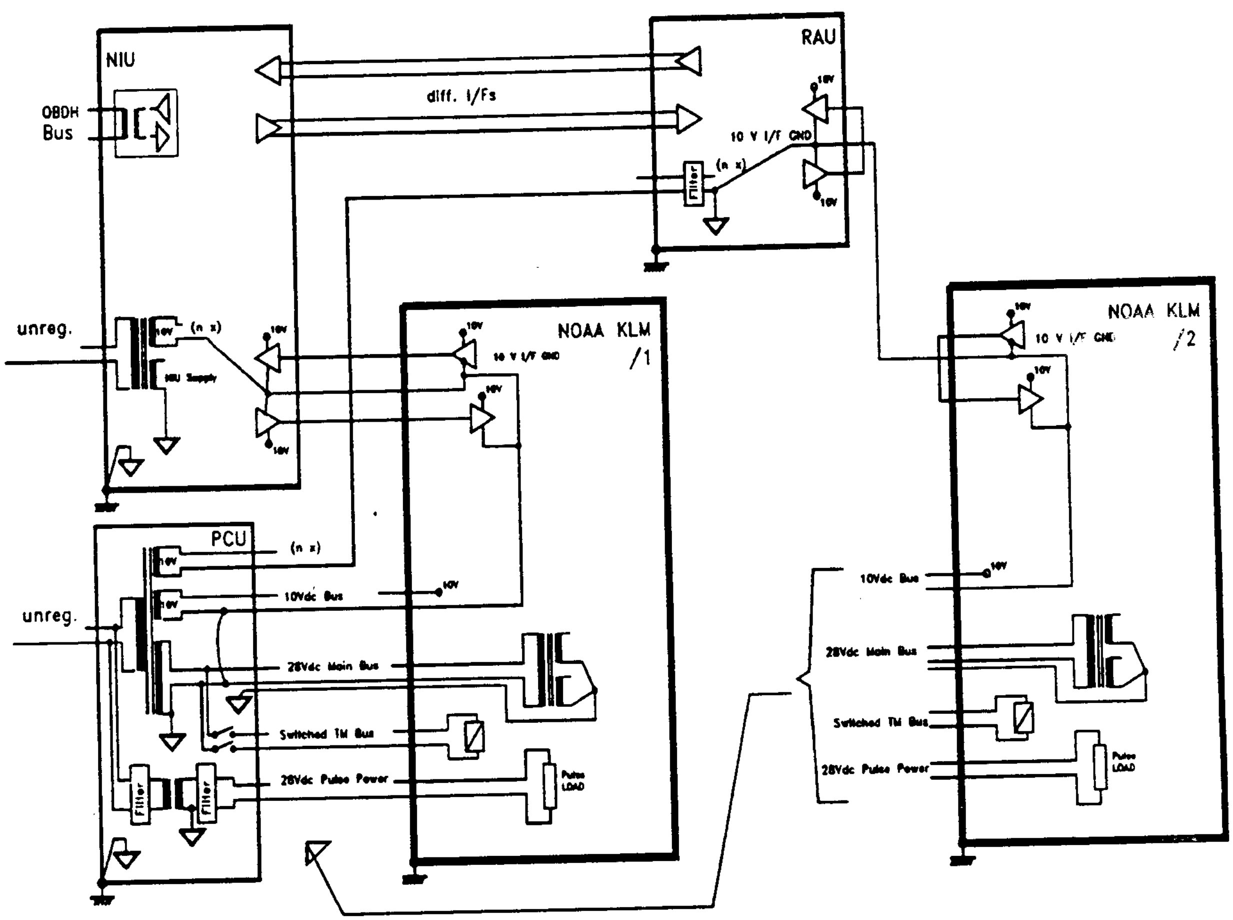


Fig. A.7.1.2- 1 Grounding and Isolation Scheme for NOAA KLM Instruments

A.7.1.2.1 Signal Interface Grounding and Isolation

Signal interfaces shall not compromise the DSPG concept. Differential receivers and single ended drivers are the recommended interfaces. Optocouplers and isolation transformers are also allowed. For the NOAA KLM instruments single ended interfaces are accepted as far as the interface line length can be minimized.

A.7.1.2.2 Spacecraft Unregulated Power Line Isolation

At unregulated power loads side (instrument input resp. converter box, PCU, input) each of the 28 V Unregulated Power buses shall be galvanically isolated from the ground reference structure and from any other power within the power load (instrument resp. converter box). The isolation shall be at least 1 MΩ shunted by not more than 50 nF between:

- primary power lead and chassis
- primary power return lead and chassis
- primary power return and secondary power return

A.7.1.2.3 Regulated Power Power Line Isolation (NOAA KLM instruments)

The particular regulated power inputs of the NOAA KLM instruments shall be galvanically isolated from the ground reference structure and from any other power within the instruments. The isolation between:

- primary power lead and chassis
- primary power return lead and chassis
- primary power return and secondary power return

shall be at least:

100 kΩ shunted by not more than 50 nF (tbc)

A.7.1.2.4 Secondary Power Grounding and Isolation

With exception of the KLM instruments the internally generated secondary voltages shall be grounded and referenced to ground structure as close as possible to the converter where generated. At the location where the secondary power is referenced to structure the maximum resistance between power return line and structure shall be ≤ 2.5 mΩ (not applicable for NOAA KLM instruments). With the exception of the secondary single point ground reference the galvanic isolation between

- secondary power lead and chassis
- secondary power return lead and chassis
- secondary power return and primary power return

shall be at least:

≥ 1 MΩ shunted by ≤ 50 nF in general

≥ 100 kΩ shunted by ≤ 50 nF (tbc) for NOAA KLM instruments

A.7.1.3 Shielding

Each electronic equipment (excluding units with intentional apertures such as openings for sensors) shall be enclosed in a electrically conductive and non-mangnetic metallic case which shall form an all-enclosing shield for electromagnetic fields. Venting holes shall be designed to provide electromagnetic shielding of better than 60 dB in the frequency range 1 GHz - 40 GHz.

A.7.1.4 Inrush Current

The individual inrush current characteristics of the NOAA instruments are specified in the relevant specifications:

NOAA KLM	RCA IS-3267415, Oct. 91	ATN-KLM General Instrument Interface Specification
NOAA 2000	GSFC-S-480-53, Aug. 93	Instrument Interface Description for NOAA 2000 Instruments with European Morning S/C and/or NOAA O,P&Q S/C

For the EDI instrument the inrush current (at instrument switch-on) shall have the following characteristics:

peak inrush current:	I < 20 Ampere in max. 20 µs	for instruments with normal operating current of 5 A < In < 20 A
- " -	I < 4 * In	for instruments with normal operating current of In <5 A
max. duration of peak inrush current:	20 µs	
max rise/fall time of current:	$dI/dt < 2 \times 10^6$ A/s	

The time duration of the complete inrush current transient envelope shall not exceed 50 ms, after which the nominal operation current shall be reached.

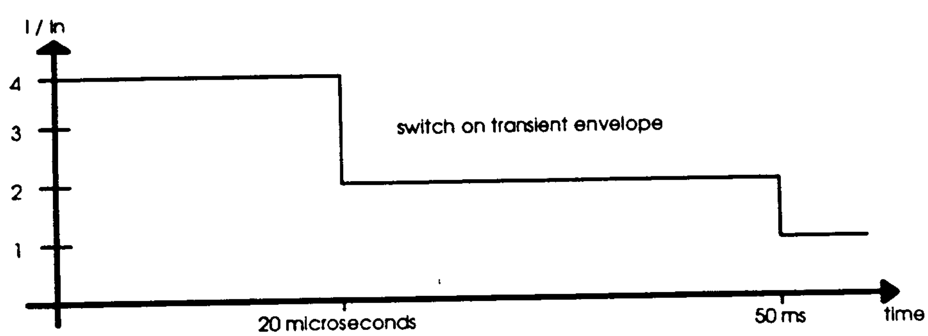


Fig. A.7.1.4-1: Switch-On Transient Envelope

A.7.1.5 Frequency List

TBD

A.7.2 EMC Performance

A.7.2.1 Power Quality

Interface requirements between the Converter Box (PCU) and the NOAA instruments regulated power input according to RCA IS-3267415, Oct. 91, ATN-KLM General Instrument Interface Specification.

A.7.2.2 Conducted Emissions

Interface requirements between the Converter Box (PCU) and the NOAA instruments regulated power input will be according to RCA IS-3267415, Oct. 91, ATN-KLM General Instrument Interface Specification. Dedicated requirements will be established later after analysis of NOAA instruments to PCU interactions.

A.7.2.3 Conducted Susceptibility

Interface requirements between the Converter Box (PCU) and the NOAA instruments regulated power input according to RCA IS-3267415, Oct. 91, ATN-KLM General Instrument Interface Specification.

A.7.2.4 Radiated Emissions

Each particular instrument shall not radiate electric fields (to be narrowband measured) in the frequency range 10 kHz - 18 GHz in excess of the limits shown in fig. A.7.2.4-1, measured in 1 metre distance. For RF and mm-wave instruments the upper frequency limit is 40 GHz (instead of 18 GHz).

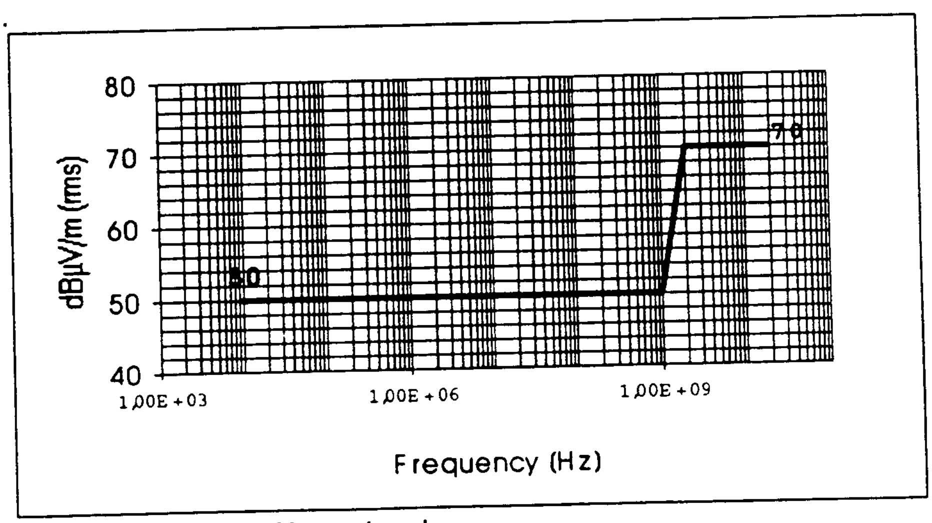


Fig. A.7.2.4-1 RE Electric Field, Narrowband

A.7.2.5 Radiated Susceptibility

Each instrument shall perform within specification accepting the test levels as specified in column 2 [required]:

Frequency	RS-E-Field Level [required]	Comment
10 kHz ...1 GHz	6 Vrms/m	general limit, derived from ENVISAT and ERS-1
1 GHz ... 18 GHz	20 Vrms/m	general limit
18 GHz ... 40 GHz	20 Vrms/m	general limit for RF and mm-wave equipment
1.6GHz ... 1.8GHz	20 Vrms/m	HRPT spot frequencies
5.1 ... 5.4 GHz	80 Vrms/m	ASCAT spot frequencies
7 GHz... 8 GHz	20 Vrms/m	XBS spot frequencies

Table A.7.2.5-1: RS Test Level Requirements

A.7.2.6 Electrostatic Discharge (ESD)

Although METOP is to be flown in polar Orbit electrostatic charging can built up especially since no conductive coating is applied for MLI. In order to prove insensitivity against ESD each particular instrument and equipment shall be subjected to an suitable test. Each instrument or equipment shall be exposed to a repetitive discharge of at least 10 mJ (tbd) and a voltage of 10 / 12 / 15 kV (tbd) without showing any deviation from specified performance. The repetition rate shall be 1/s and the test duration > 3 min.

A.8 RFC Design

The RFI model below indicates the RF transmitter, the coupling path which shall contain both the transmitting and the receiving antenna and the RF receiver.

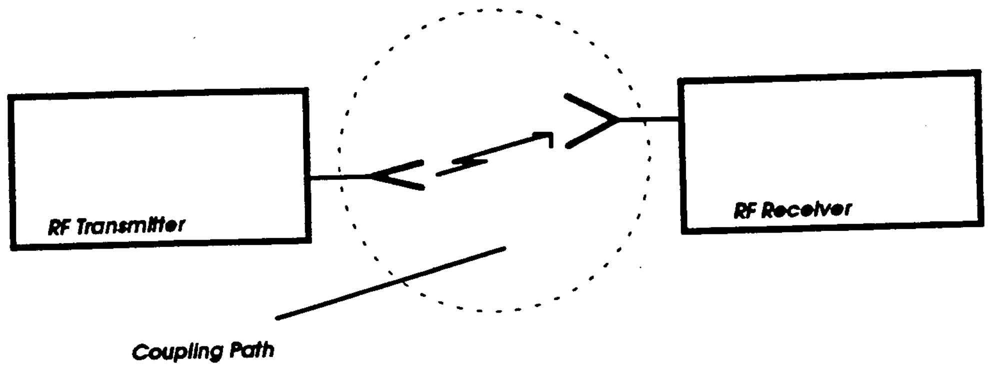


Fig. 4-1 General Interference Model

The RF transmitter is characterized by parameters such as frequency, output power, bandwidth and spurious/harmonics. The coupling path will be characterized by antenna coupling factors. The antenna coupling factor describes the interference coupling from the input port of the transmitting antenna to the output port of the receiving antenna and therefore it includes propagation losses over the radiation path, diffraction around obstructions, reflections from planar surfaces and antenna gain factors. The RF receiver itself is principally characterized by its sensitivity across frequency range (i.e. input filter bandwidth).

A.8.1 RF Transmitter Characteristics

In general the transmitter characteristics can be derived by considering the transmitter architecture as outlined in fig. A.8.1-1. The band limited High Power Amplifier (HPA) provides for amplification of the transmitting signals. In most cases the HPA is considered to be the main source for unwanted spurious and harmonics generation. Therefore the transmitted signal must be bandpass filtered at the output of the HPA before it reaches the antenna. For high power transmitters such as the ASCAT transmitter the waveguide, mandatory at the output of the transmitter, will introduce additional high-pass filtering of the transmitting signals.

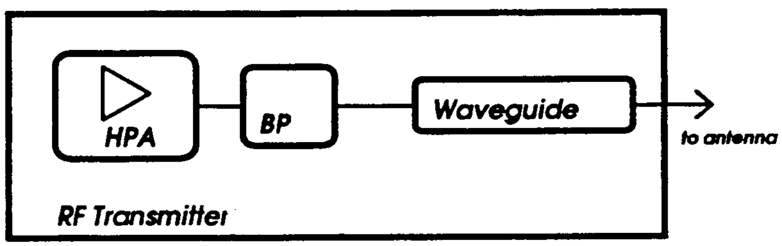


Fig. A.8.1-1: RF Transmitter Model

The RF transmitters have the following specified characteristics:

	CF	Pout	BW	Mod	PW	PRF	Comments
ASCAT	5.255 GHz	50.8 dBm	124 kHz 411 kHz	12 kHz/ms 50 kHz/ms	10.32 ms 8.22 ms	29.63 Hz	side beam mid beam
none	x	x	x	x	x	x	x
LRPT	137.1 MHz (137.9 MHz)	38 dBm	150 kHz (150 kHz)	BPSK (BPSK)		CW (CW)	backup
HRPT	1.707 GHz (1.701 GHz)	42 dBm	4.5 MHz (4.5 MHz)	QPSK (QPSK)		CW (CW)	backup
SBS	2206 MHz	23 dBm	400 kHz	PSK		1.5 rad	
XBS	7.5 GHz 7.8 GHz	44 dBm	375 MHz	BPSK/QPSK		50/100 Mbps	option

Table A.8.1-1: RF Transmitters, Specified Parameters

For the RFI analysis each particular transmitter shall have a output spectrum as described in fig. A.8.1-2 and table A.8.1-2. The levels of the harmonics as generated by the particular transmitter are listed in table A.8.1-3.

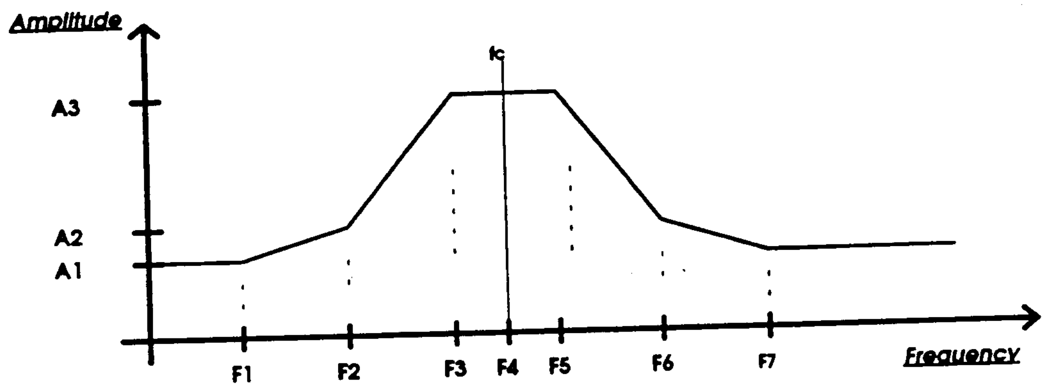


Fig. A.8.1-2: Transmitter Output Spectrum Model

	ASCAT	none	LRPT	HRPT	SBS	XBS 7.5 GHz
F1	5.155 GHz	x	136.98 MHz	1.6995 GHz	2205.4 MHz	7.0 GHz
F2	5.254 GHz	x	137.04 MHz	1.70325 GHz	2205.8 MHz	7.15 GHz
F3	5.2547 GHz	x	137.08 MHz	1.70475 GHz	2205.8 MHz	7.3 GHz
F4	**5.255 GHz**	x	**137.1 MHz**	**1.707 GHz**	**2206 MHz**	**7.5 GHz**
F5	5.2553 GHz	x	137.12 MHz	1.70925 GHz	2206.2 MHz	7.7 GHz
F6	5.256 GHz	x	137.16 MHz	1.71075 GHz	2206.2 MHz	7.75 GHz
F7	5.355 GHz	x	137.22 MHz	1.7145 GHz	2206.6 MHz	7.8 GHz
F5 - F3 A3	600 kHz 50.8 dBm	x	40 kHz 38 dBm	4.5 MHz 42 dBm	400 kHz 23 dBm	400 MHz 44 dBm
F6 - F2 A2	2.0 MHz 20.8 dBm	x	120 kHz -10 dBm	7.5 MHz -10 dBm	400 kHz 13 dBm	600 MHz 34 dBm
F7 - F1 A1	200 MHz -19.2 dBm	x	240 kHz -20 dBm	15 MHz -30 dBm	1200 kHz -37 dBm	800 MHz -15 dBm

Table A.8.1-1: Transmitters, Output Spectrum

	fc (fundamental)	2 * fc (1st harmonic)	3 * fc (2nd harmonic)	4 * fc (3rd harmonic)	5 * fc (4th harmonic)	6 * fc (5th harmonic)	7 * fc (6th harmonic)
ASCAT	0 dBc	-40 dBc	-60 dBc	-70 dBc	-80 dBc	-90 dBc	-100 dBc
none	x	x	x	x	x	x	x
LRPT	0 dBc	-40 dBc	-60 dBc	-70 dBc	-80 dBc	-90 dBc	-100 dBc
HRPT	0 dBc	-40 dBc	-60 dBc	-70 dBc	-80 dBc	-90 dBc	-100 dBc
SBS	0 dBc	-40 dBc	-60 dBc	-70 dBc	-80 dBc	-90 dBc	-100 dBc
XBS	0 dBc	-50 dBc	-50 dBc	-70 dBc	-80 dBc	-90 dBc	-100 dBc

Table A.8.1-3: Transmitters, Harmonic Levels

A.8.2 Receiver Characteristics

In general the receiver characteristics can be derived from fig. A.8.2-1. The sensitivity of the receiver is given by both the sensitivity of the Low Noise Amplifier (LNA) and additional losses within the path between antenna output port and LNA input. The receiving frequency band is mainly limited by the receiver input bandpass filters (discrete bandpass and additional waveguide with high-pass character) in front of the LNA.

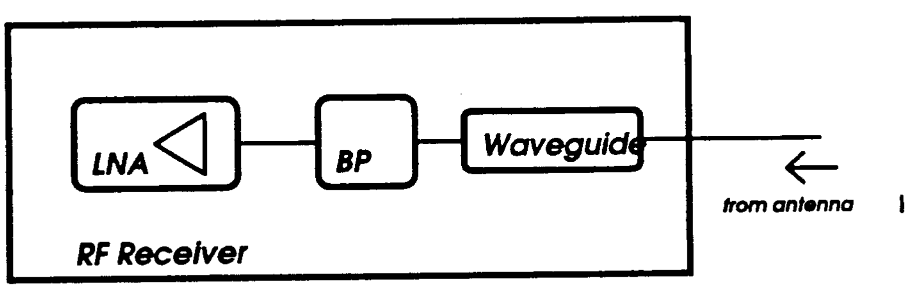


Fig. A.8.2-1: RF Receiver Model

The receivers as considered for RFI analysis have the following specified characteristics:

	CF	BW	Sensitivity	Comments
SBS	2021.3 MHz	400 kHz	-140 dBm	
DCS	401.65 MHz	160 kHz	-154 dBm	via UDA
ASCAT	5.255 GHz	600 kHz	-154 dBm	
MIMR	6.8 GHz 10.65 GHz 18.70 GHz 23.80 GHz 36.50 GHz 89.0 GHz	200 MHz 100 MHz 200 MHz 400 MHz 1000 MHz 5400 MHz	-129 dBm -130 dBm -125 dBm -122 dBm -119 dBm -112 dBm	
AMSU A1 A2 A3	 23.8 GHz 31.4 GHz 50.3 GHz	 275 MHz 180 MHz 180 MHz	 -94.8 dBm -96.7 dBm -96.7 dBm	

Table A.8.2-1: RF Receivers, Specified Parameters

For the RFI analysis each particular receiver shall have the sensitivity curve over frequency as described in fig. A.8.2-2 and table A.8.2-2a-b.

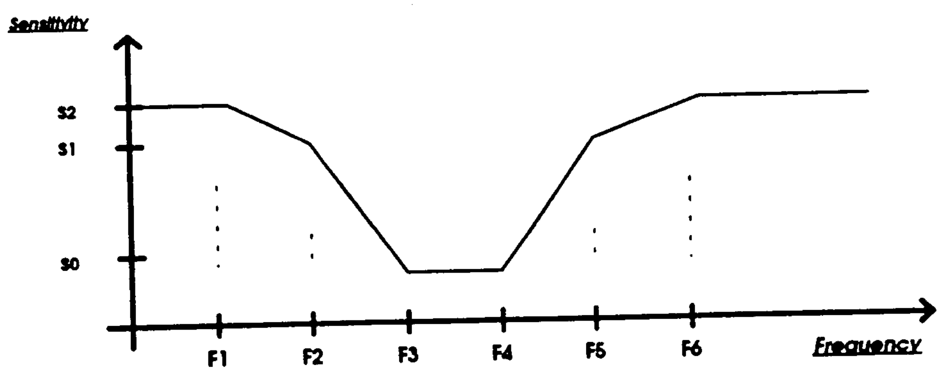


Fig. A.8.2-2: Receiver Sensitivity Model

	SBS	DCS 401.65 MHz	ASCAT 5.255 GHz	AMSU A1	AMSU A2	AMSU A3
F1	1921.3 MHz	375 MHz	5.155 GHz	21.42 GHz	28.26 GHz	45.27 GHz
F2	2011.3 MHz	385 MHz	5.245 GHz	23.63 GHz	31.29 GHz	50.19 GHz
F3	2019.8 MHz	401.57 MHz	5.2547 GHz	23.66 GHz	31.31 GHz	50.21 GHz
F4	2022.8 MHz	401.73 MHz	5.2553 GHz	23.935 GHz	31.49 GHz	50.39 GHz
F5	2031.3 MHz	411 MHz	5.265 GHz	23.962 GHz	31.508 GHz	50.408 GHz
F6	2121.3 MHz	425 MHz	5.355 GHz	26.18 GHz	34.54 GHz	55.33 GHz
F4 - F3 S0	3 MHz -140 dBm	160 kHz -154 dBm	600 kHz -154 dBm	275 MHz -94.8 dBm	180 MHz -96.7 dBm	180 MHz -96.7 dBm
F5 - F2 S1	20 MHz S0 + 60 dB	26 MHz S0 + 25 dB	20 MHz S0 + 30 dB	332 MHz S0 + 40 dB	200 MHz S0 + 40 dB	218 MHz S0 + 40 dB
F6 - F1 S2	200 MHz S0 + 140 dB	50 MHz S0 + 130 dB	200 MHz S0 + 70 dB	4.76 GHz S0 + 40 dB	6.28 GHz S0 + 40 dB	10.06 GHz S0 + 40 dB

Table A.8.2-2a: Receiver Sensitivity

	MIMR 6.8 GHz	MIMR 10.65 GHz	MIMR 18.70 GHz	MIMR 23.80 GHz	MIMR 36.50 GHz	MIMR 89.0 GHz
F1	6.5 GHz	10.5 GHz	18.4 GHz	23.2 GHz	35.0 GHz	80.9 GHz
F2	6.67 GHz	10.58 GHz	18.57 GHz	23.54 GHz	35.85 GHz	85.49 GHz
F3	6.7 GHz	10.6 GHz	18.6 GHz	23.6 GHz	36.0 GHz	86.3 GHz
F4	6.9 GHz	10.7 GHz	18.8 GHz	24.0 GHz	37.0 GHz	91.7 GHz
F5	6.93 GHz	10.72 GHz	18.83 GHz	24.06 GHz	37.15 GHz	92.51 GHz
F6	7.1 GHz	10.8 GHz	19.0 GHz	24.4 GHz	38.0 GHz	97.1 GHz
F4 - F3 S0	200 MHz -129 dBm	100 MHz -130 dBm	200 MHz -125 dBm	400 MHz -122 dBm	1000 MHz -119 dBm	5400 MHz -112 dBm
F5 - F2 S1	260 MHz S0+40dB	140 MHz S0+40dB	260 MHz S0+40dB	520 MHz S0+40dB	1300 MHz S0+40dB	7020 MHz S0 + 40dB
F6 - F1 S2	600 MHz S0+60dB	300 MHz S0+60dB	600 MHz S0+60dB	1200 MHz S0+60dB	3000 MHz S0+60dB	16200 MHz S0+60dB

Table A.8.2-2b: Receiver Sensitivity

A.9 EGSE

A.9.1 EGSE Overview

A.9.2 EGSE Interfaces

A.9.3 EGSE to Instrument Interfaces

A.9.4 EGSE Infrastructure / Facilities Requirements

A.9.5 EGSE Handling / Transportation / Storage

A.10 Ground Operations

B. EMI, EDI, AND AO INSTRUMENT INTERFACE CHARACTERISTICS

B.1 Introduction

The general avionic interfaces of this group of instruments comply in general with the common avionics interfaces for the instruments on PPF/ENVISAT. As a consequence of that the relevant PPF/ENVISAT requirements are in general applicable and could be consulted for further details.

B.2 Overview

The main differences to ENVISAT appear in the area of power distribution where the individual switching of power busses per instrument on PLM side is replaced by a system of commonly switched power busses to reduce the number of relays on PLM side and hence simplify the overall PLM avionics architecture.
Similar has been implemented for Thermal Control where now only one TCU is replacing the two Heater Switching Units needed on ENVISAT, however, this does not impact the instrument interfaces.

Command and Control of this group of instruments is characterized by the delegation of instrument command and control to a dedicated Instrument Control Unit (ICU) which is connected to the PLM OBDH bus and follows the standard high level protocol of ERS and ENVISAT.

For the baseline of the Phase A it has been agreed with EUMETSAT and ESA that also IASI and MHS form part of this group of instruments. It is understood that EUMETSAT has actions pending to confirm this assumption for further phases or otherwise.
The overall PLM architecture is shown in Figure B.2-1.

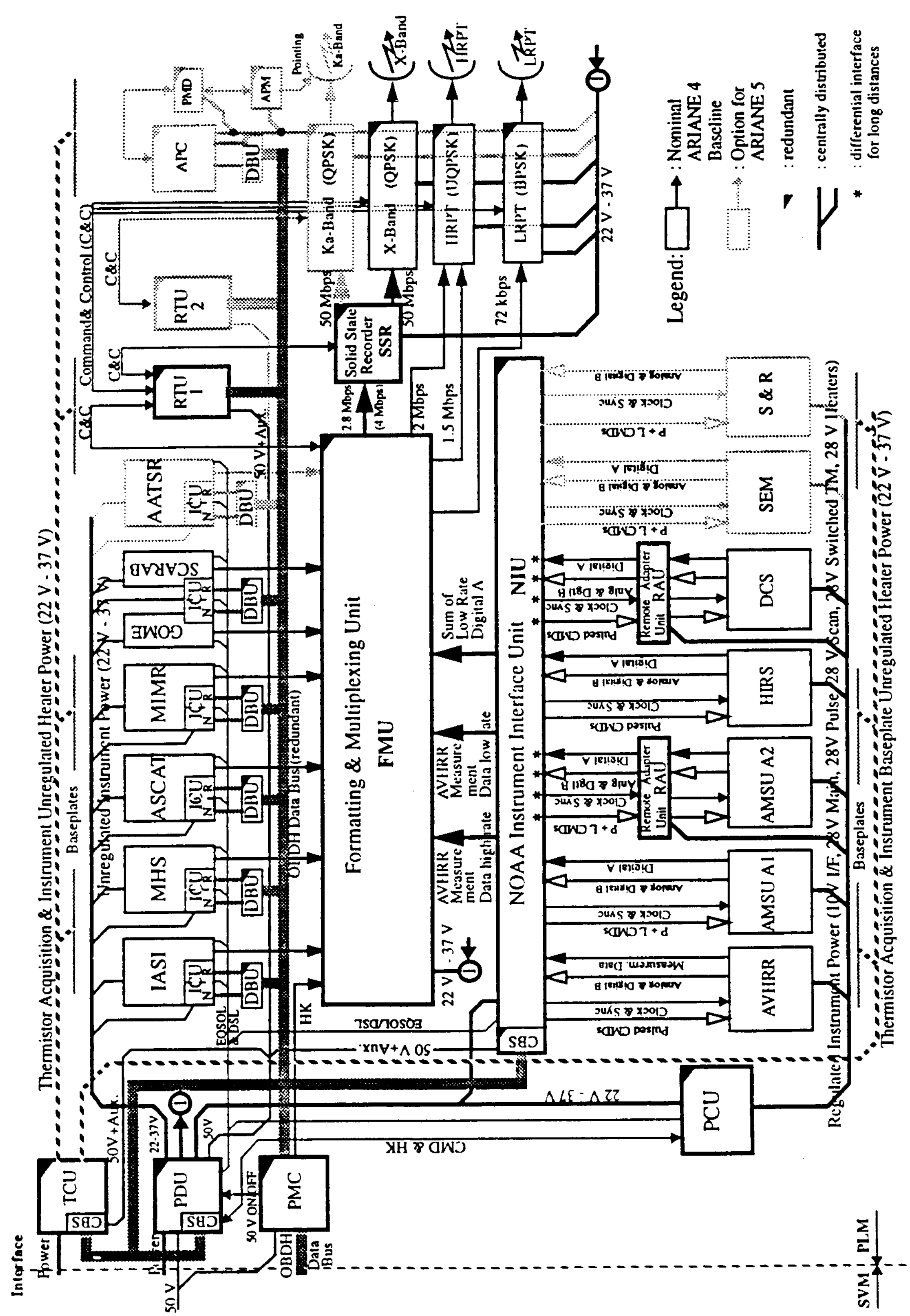


Figure B.2-1: Overall PLM Architecture

B.3 Electrical Power Interfaces

The EMI, EDI, and A.O. instruments will be provided with 4 types of redundant power interfaces by the PLM.

- A +22 to +37 Volt unregulated Equipment Power Bus as primary source for the instrument equipment, which is commonly switched and protected by a current limiter in the PDU.
- A +22 to +37 Volt unregulated ICU Power Bus which is individually switched and protected in the PDU per instrument ICU.
- A +22 to +37 Volt unregulated Heater Power Bus for externally located instrument equipment to maintain non-operating temperatures when the instrument ICU is not operating.
- A +22 to +37 Volt unregulated ICU Heater Power Bus for externally located instrument ICU's to maintain them on operating temperatures when the ICU is not operating.

The availibility of these busses in the various PLM modes is shown in table B.3-1.

Modes	Inert	Ground	LEOP	Stdby	Operable		Fix	Safe
Power					Inst. Op.	Inst. Off		
EQU Power Bus	off	any	off	on	on	on	off	off
ICU Power Bus	off	any	off	any	on	off	off	off
EQU Heater Bus	off	any	off->on	any 1)	off	on	on	on
ICU Heater Bus	off	any	off->on	any 1)	off	on	on	on

off->on = switched on by ground command after successful solar array deployment and when necessary from thermal point of view

1) = off if ICU on and vice versa

Table B.3.-1: Power Bus Availibility

The voltage range and profile per orbit at all 4 bus types is defined in figure B.3-1.

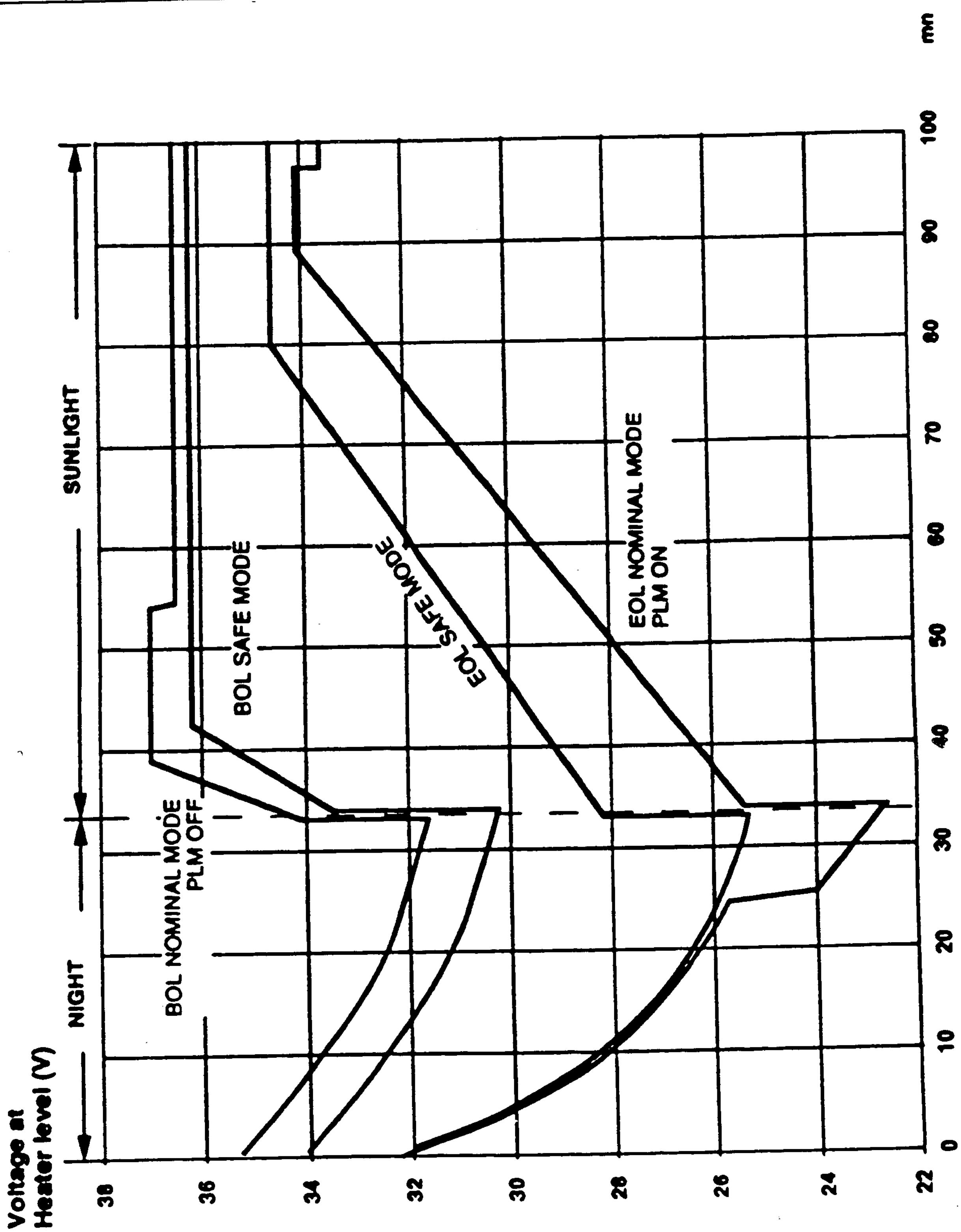


Figure B.3-1: Unregulated Power Buses - Voltage Profile

B.4 Command & Control

B.4.1 OBDH Bus Protocol

Instruments with ICU's shall communicate with the PMC via a standardized 3 level protocol specified for ENVISAT and derived from the ERS protocol.
The OBDH Bus Protocol is specified in detail in ID27, ENVISAT OBDH and Measurement Data Protocol Specification PO-RS-DOR-PM-0010.

B.4.2 General Macrocommands

General Macrocommands are specified in ID28, ENVISAT Command & Control Specification PO-RS-DOR-PL-0007 and will be further detailed or adapted at a later stage.

B.4.3 ICU Housekeeping Formats

ICU Housekeeing Formats are in general specified in ID27, ENVISAT OBDH and Measurement Data Protocol Specification PO-RS-DOR-PM-0010.

Detailed requirements on the contents and ICU internal handling are given in ID28, ENVISAT Command & Control Specification PO-RS-DOR-PL-0007.

ICU Dumps are specified in ID27, ENVISAT OBDH and Measurement Data Protocol Specification PO-RS-DOR-PM-0010.

B.4.4 Satellite Services

The PMC provides some special functions which instruments may use for their own purposes. The following subchapters shall in future identify the relevant functions individually.

B.4.4.1 General Macrocommands

General Macrocommands are defined in ID28.

B.4.4.2 Housekeeping Formats

Housekeeping Format requirements are defined in ID28.

B.4.4.3 Orbit State Vector

The orbit state vector from the SVM is continuously merged into the measurement data streams by the PLM. Instruments will therefore only be provided with this vector, if they require this information for on-board purposes, e.g. sampling window adjustment etc.

B.4.4.4 Attitude State Vector

The attitude state vector from the SVM is continuously merged into the measurement data streams by the PLM. Instruments will therefore only be provided with this vector, of they require this information for on-board purposes, e.g. sampling window adjustment etc.

B.4.4.5 Auxiliary Satellite Housekeeping Data

Auxiliary satellite housekeeping data is continuously merged into the measurement data streams by the PLM. Instruments will therefore only be provided with this kind of information if they require it for on-board purposes.

B.4.4.6 Synchronization

Instruments shall be synchronized to the satellite clock. The frequency for updating the clock is distributed by the OBDH bus frequency. Details are specified in ID27 and ID28.

B.4.4.7 Datation

Instruments shall date their measurement data as required to achieve the mission objective. Instruments shall as well date their housekeeping data to allow timely correlation with other instrument, PLM, or satellite events. Details are specified in ID27 and ID28.

B.4.5 Non-Nominal Operation

In cases where nominal operation cannot be continued anymore, the PLM will switch instruments down. Switch-down can be accomplished in several ways:

- by nominal commanding using the Standby-Macrocommand and subsequent power down of the ICU

- by requesting the instrument to power down all instrument equipment with the generation of the Equipment Switch Of Line (EQU SOL).
  This is used in case that nominal communication with the ICU is no more possible. In case of an assumed ICU failure the ICU will remain powered to maintain ist memory contents for troubleshooting, in case the failure is on PLM or satellite side the ICU will subsequently be powered down as well.

- by requesting those instrument, which might suffer from non-nominal pointing, to transfer into a protection mode (safe mode), which makes them unsusceptible to undefined sun illumination. After reaching the protection mode the PLM will power down the instrument.

Details will be specified in ID28 at a later stage. For the time being reference is made to RD10a as concerns the detailed switch down sequence in case of EQU SOL and DSL.

B.5 Measurement Data

ESA Development Instruments (EDI´s) and Announcement of Opportunity Instruments (A.O.´s) shall provide their measurement data formatted in source packets. The source packet layout is specified in the following section.

B.5.1 Layout of Source Packets

The source packet shall contain one logical measurement data set derived from the sampling needs (e.g. CCD readout, scanline, etc.), which is cyclically generated by the instrument. In addition the source packet shall contain all instrument auxiliary data necessary to interpret the measurement data on ground (housekeeping data, datation, instrument mode indication, etc.).

Each source packet shall consist of four main parts:

- The packet header (length 48 bit), the data field header (variable length), the source data (variable length) and the packet error control (optional with variable length).

Packet Header

The packet header shall be subdivided into the following three fields:

- Packet identification (16 bit)

  Consisting of the version number (3 bits), type (1 bit), data field header flag (1 bit) and the application process ID (11 bit).

  The following coding conventions shall apply:

Version number:	BIN '100'
Type:	BIN '0'
Data field header flag:	BIN '1'
Application process ID:	TBS by the METOP Project

- Packet sequence control (16 bit)

  Consisting of the segmentation flags (2 bits) and the source sequence counter (14 bit).

  The segmentation flags shall be set to BIN '11' indicating that the source packet is not segmented by the instrument.

The 14 bit wrap-around source sequence counter is used to reconstruct the correct source packet sequence after transmission to ground. The counter shall be incremented by the instrument.

- Packet length ID (16 bit)

The packet length ID shall reflect the number of bytes (minus one) from the first byte of the data field header to the last byte of the source data field (or, if used, the packet error control field). This results in the following maximum length of a source packet:

Packet header	6 bytes
max. packet length	65536 bytes
total max. length	65542 bytes

Source packets shall not be filled up with dummy data. Exceptions from this requirement have to be agreed by the METOP Project.

Note: An instrument may use source packets of different length, provided the length requirement from above are met.

It is also advisable not use very large source packets due to the fact that a loss of one transfer packet on a disturbed ground link channel may invalidate the corresponding source packet.

Data Field Header

The data field header shall contain the auxiliary data.

The presence of the data field header shall be flagged in the packet header.

The data field header shall contain as a minimum:

- Length of the data field header (in bytes)
- ICU on-board time of the measurement
- instrument redundancy definition vector as contained in the REDUNDANCY DEFINITION macrocommand
- instrument mode during measurement

The layout of the data field headers is TBS.

Source Data

The source data field layout is under instrument responsibility and shall contain the measurement data generated by the instrument. The only restriction on source data fields is that length shall be an integer multiple of bytes (octets).

Packet Error Control

As an option the source packet can be protected by a packet error control sequence. The coding procedure as well as the processing itself (coding/decoding) is under the responsibility of the instrument.

A copy of the ESA standard describing the applicable source packet format layout is given in chapter B5.4.

B.5.2 Layout of CVCDU´s

The Transfer Formatting of the instrument measurement data (source packets) into the CVCDU´s will be performed in the METOP PDHT System's FMU.

The CVCDU, used for METOP shall have a uniform length of 1020 octets (bytes).

The CVCDU shall consist of the following main parts:

- The CVCDU Primary Header 8 octets (64 bit)
- The CVCDU Data Field 884 octets (7072 bit)
- The CVCDU Reed-Solomon Check Symbols 128 octets (1024 bit)

CVCDU Primary Header

The CVCDU Primary Header contains the following fields:

CVCDU Identification Field (16 bit) consisting of:

- Version Number (2 bit) shall be set to BIN '01'
- Spacecraft ID (8 bit)
- Virtual Channel ID (6 bit)

The 6 bit virtual channel ID provides up to 64 virtual channels for a particular spacecraft.

The coding of the CVCDU identification field will be specified by the METOP Project at an appropriate time.

CVCDU Counter (24 bit):

- The CVCDU counter is an 24 bit wrap around counter which counts the CVCDU´s of each of the 64 virtual channels.

CVCDU Signalling Field (8 bit) consisting of:

- Replay Flag (1 bit)
- Reserved spares (7 bit)

The replay flag shall be set to BIN '0', indicating real-time CVCDU´s.

The reserved spare bits are currently not used and shall be set to "all zeros".

<u>CVCDU Insert Zone</u>

The CVCDU insert zone (2 octets) will be used for the insertion of the encryption control information into the LRPT and HRPT service CVCDU´s.
For the Global Data CVCDU´s these 2 octets will be set to $øø_{Hex}$.

<u>CVCDU Data Field</u>

The CVCDU data field in general has a variable length but by choosing a CVCDU length. The length of the data field is also determined. For the CVCDU length of 1020 bytes the dat field is set to 884 bytes.

The CVCDU data field consists of the following parts:

- Protocol Data Unit Header (PDU-Header) 2 octets (16 bits)
- Data Unit Zone (for the 1020 octet CVCDU) 882 octets (7056 bits)

The PDU-Header consists of 5 spare bits and the First Header Pointer (11 bits). The First Header Pointer is used for localisation of the first source packet header in the CVCDU data field. If no source packet header is contained in the relevant CVCDU the pointer shall be set to all one.

The data unit zone shall contain source packets as specified in section B5.1.

Source packets exceeding 882 bytes shall be transmitted in several VCDU´s. The instrument shall not fill incomplete data unit zones with zeroes, but continue with data from the next source packet.

CVCDU Reed-Solomon Check Symbols field

The Reed-Solomon check symbol field shall be 128 bytes. This corresponds to a protected information length of 884 bytes + 8 bytes = 892 bytes for the used Reed-Solomon code.

VCDU Identification (VCDU-ID)			Vitual Cannel Data Unit Counter	Signaling Field		VCDU Insert Zone	Protocol Data Unit Header		CVCDU Data Unit Zone	CVCDU Reed Solomon Check Symbols
Version Number	S/C-ID	Virtual CH. ID		Replay Flag	Spare		Spare	First Header Pointer		
2 bits	8 bits	6 bits	24 bits	1 bit	7 bits	16 bits	5 bits	11 bits		
2 Octets			3 Octets	1 Octet		2 Octets	2 Octets		882 Octets	128 Octets

Figure B.5.2-1 CVCDU layout

B.5.3 Low/Medium Data Interface

The Low/Medium Rate Data Interface shall be used by instruments whose data rate will not exceed 32 Mbps.

The data transfer on the low/medium data interface shall be performed on a serial interface that provides functional cross-strapping within the instrument between redundant instrument units and a redundant FMU.

Figure B.5.3-1 shows the principle interface layout and in figure B.5.3-2 and B.5.3-3 the interface timing is described.

Instrument units involved in the data transfer shall only be powered in instrument modes which imply data transfer. When no data transfer is required the packet valid flag shall stay reset.

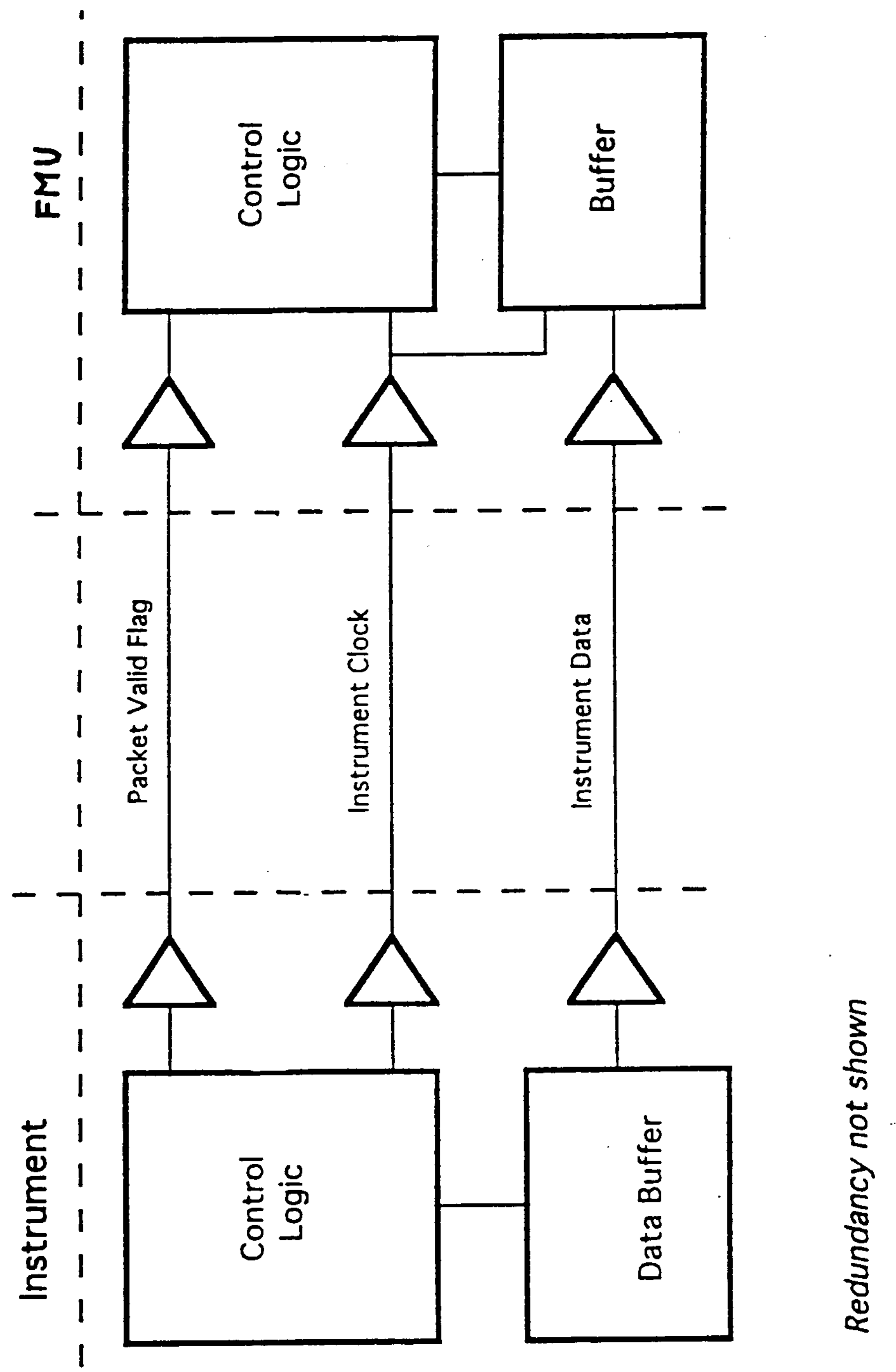


Figure B.5.3-1 Blockdiagram of Payload Data Interface

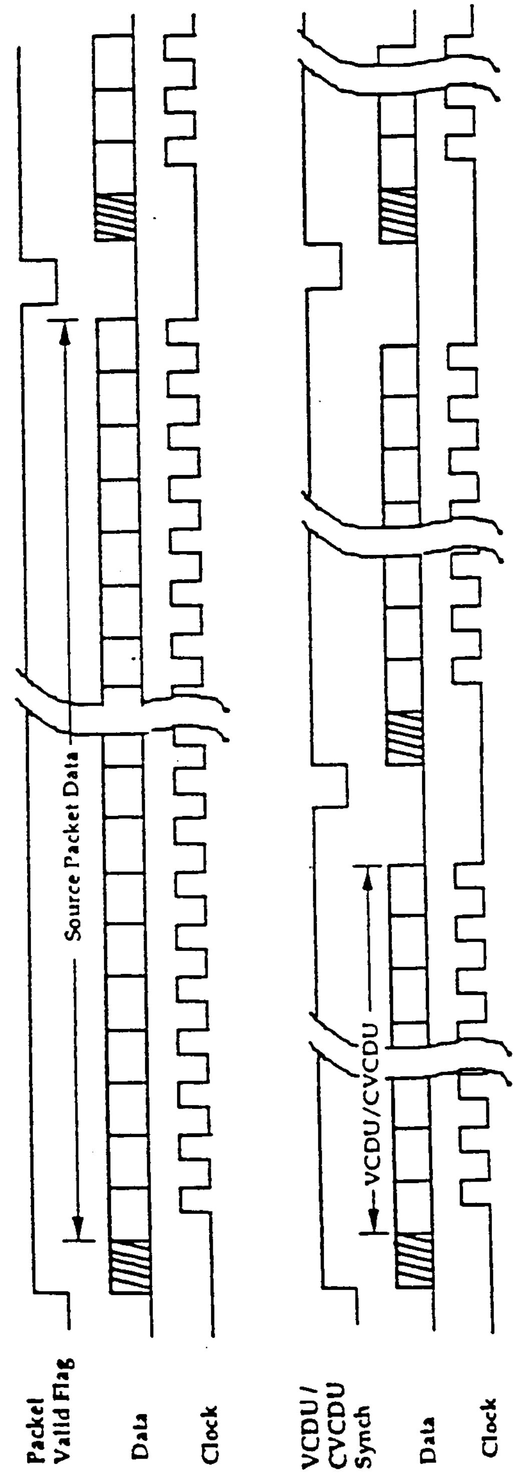


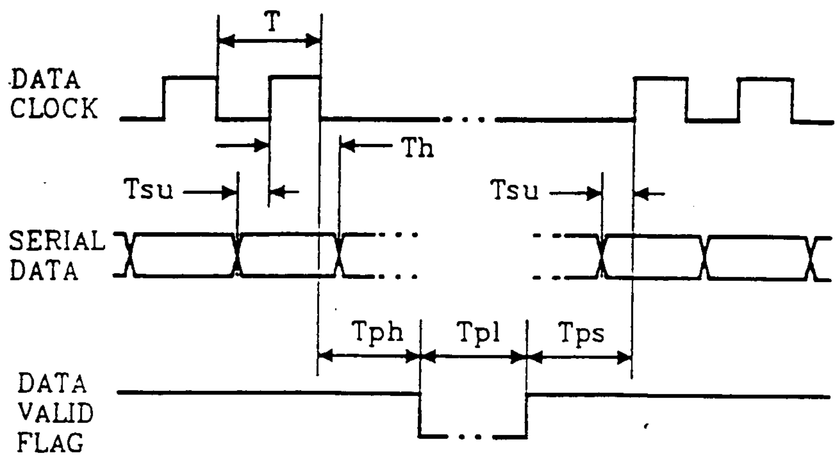
Figure B.5.3-2LBR Timing Schematics

The initialisation of a data transfer shall depend on the instrument flagging "Packet Valid". This signal shall remain set for the complete source packet. Subsequently after the packet valid flag setting, the instrument shall send the source packet data with an associated clock to the HSM.

The termination of the data transfer to the HSM shall be controlled by the instrument (reset of packet valid flag). This shall occur at the end of a complete source packet.

Between tow consecutive source packets the instrument shall reset the Packet Valid Flag.

The instrument dataline shall stay reset if no data are transferred.



TIME	DESCRIPTION	MAX	MIN
T	DATA CLOCK PERIOD		100 ns
Tsu	DATA VALID BEFORE CLOCK EDGE		40 ns
Th	DATA VALID AFTER CLOCK EDGE		10 ns
Tph	VALID FLAG HIGH AFTER LAST CLOCK	T + 30 ns	T
Tpl	DURATION OF VALID FLAG LOW		T
Tps	VALID FLAG HIGH BEFORE FIRST CLOCK	T + 30 ns	T
Tr	RISE/FALL TIME - DATA AND CLOCK	15 ns (TBC)	
Tji	PHASE JITTER - DATA TO CLOCK	± 3 ns	

Figure B.5.3-3 LBR Timing

The Low/Medium data rate interface shall have the electrical characteristics as defined in chapter B.6.4.

B 5.4 Source Packet Definition (according to ESA Standard)

B 5.4.1 General

This section defines one protocol data unit:

- the protocol data unit which must be presented by the spaceborne data source to the data transfer system for transmission to the ground: This unit is the SOURCE PACKET.

The source Packet must, in addition to the source data, carry a minimum of information needed by the ground data capture system for the acquisition, storage and distribution of the source data to the end user. Thus, the Source Packet format consists of two major fields:

- The Packet Header, of fixed length, which provides the standardised control information required during the end-to-end transport process from the source on board the spacecraft to the end-user data processing equipment on the ground.
- The Packet Data Field, of variable length, which contains the source data.

The standardised control information that the Source Packet Header must provide is the following:

- Identification of the source and its application process: For data distribution, storage and retrieval.
- Sequence numbering for a given source and its application process: For sequence tracking and accounting.
- Packet Data Field length: Information used throughout the transport process.

The only other constraint placed od the data source is that the length of the Packet Data Field must not exceed 2^{16} (65536) octets.
The Source Packet format is specified in Section B.5.4.2.

Packet Header (48)							Packet Data Field (Variable)		
Packet Identification				Packet Sequence Control		Packet Length	Data Field Header (Optional) May contain: - S/C Time - Packet Format -Ancillary Data	Source Data	Packet Error Control (Optional)
Verions Number 3	Type 1	Data Field Header Flag 1	Application Process ID 11	Segmentation Flags 2	Source Sequence Count 14				
16				16		16	Variable	Variable	Variable

Figure B 5.4.1 SOURCE PACKET FORMAT

B 5.4.2 Source Packet Format

The Source Packet Format, which is shown in Figure B 5.4.1, consists of two major fields:

- The Packet Header, with a fixed length of 48 bits (6 octets).
- The Packet Data Field, with a variable length from 1 to 65 536 (2^{16}) octets.

B 5.4.2.1 Source Packet Header

The Packet Header shall consist of 48 bits subdivided into the following fields:

FIELD	LENGTH (BITS)
PACKET IDENTIFICATION	
- Version Number (3)	16
- Type (1)	
- Data Field Header Flag (1)	
- Application Process ID (11)	
PACKET SEQUENCE CONTROL	16
- Segmentation Flags (2)	
- Source Sequence Count (14)	
PACKET LENGTH	16
	48

B 5.4.2.1.1 Packet Identification (16 Bits)

Packet Identification is a 16-bit field divided into four subfields namely Version Number (3 bits), Reserved Bit (1 bit), Data Field Header Flag (1 bit) and Application Process Identifier (11 bits).

***(a) Version Number** (bits 0 through 2)*

The Version number is a 3-bit field occupying the three most significant bits of a packet structure. The Version Numbers are defined by the CCSDS. IN THIS STANDARD, ONLY ONE VERSION NUMBER (VERSION 2) IS PERMITTED, and this specifies the packet formats described in this Section. This number is:

- Bits 0 through 2 = 100

(b) Type (Bit 3)

The Data Field Header Flag indicates the presence (Bit 4 = 1) or absence (Bit 4 = 0) of a Data Field Header within the Packets shall have this bit set to '0'.

(c) Data Field Header Flag (Bit 4)

The Data Field Header Flag indicates the presence (Bit 4 = 1) or absence (Bit 4 = 0) of a Data field Header within the Packet Data Field.

B 5.4.2.1 Source Packet Header

The Packet Header shall consist of 48 bits subdivided into the following fields:

FIELD	LENGTH (BITS)
PACKET IDENTIFICATION	
- Version Number (3)	16
- Type (1)	
- Data Field Header Flag (1)	
- Application Process ID (11)	
PACKET SEQUENCE CONTROL	16
- Segmentation Flags (2)	
- Source Sequence Count (14)	
PACKET LENGTH	16
	48

B 5.4.2.1.1 Packet Identification (16 Bits)

Packet Identification is a 16-bit field divided into four subfields namely Version Number (3 bits), Reserved Bit (1 bit), Data Field Header Flag (1 bit) and Application Process Identifier (11 bits).

*(a) **Version Number** (bits 0 through 2)*

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- Bits 0 through 2 = 100

(b) Type (Bit 3)

The Data Field Header Flag indicates the presence (Bit 4 = 1) or absence (Bit 4 = 0) of a Data Field Header within the Packets shall have this bit set to '0'.

(c) Data Field Header Flag (Bit 4)

The Data Field Header Flag indicates the presence (Bit 4 = 1) or absence (Bit 4 = 0) of a Data field Header within the Packet Data Field.

(d) Application Process Identifier (Bits 5 through 15)

The Application Process Identifier is an 11-bit field uniquely identifying both the physical source (instrument or subsystem unit) and the particular application process within this physical source which created the Source Packet. A physical source may 'own' more than one application process. Any identifier is unique on board a given spacecraft, regardless of the number of Virtual Channels used.

The Application Process Identifiers are tailored to the mission needs, in general, and to the overall data handling system requirements, in particular. They are ultimately assigned by the Mission Control authority. Each Application Process Identifier is logically associated with the Source Sequence Count subfield of the Packet Sequence control field. This is to allow the ground telemetry acquisition systems to control the continuity of packet delivery for each Application ID (see Subsection B 5.4.2.1.2).

Reserved Identifiers

- The "all ones" configuration of the Application Process ID shall be reserved to identify "Idle Source Packets" which are generated by the spacecraft data transport system to maintain synchronisation of the packet extraction process on ground during periods when no sources have packetised data available for transfer to the ground.
- The "all zeros" configuration of the Application Process ID shall be reserved to identify the standard Source Packet containing the Spacecraft Time sample.

B 5.4.2.1.2 Packet Sequence Control (16 Bits)

Packet Sequence Control is a 16-bit field which is subdivided into two separate fields, namely Segmentation Flags (2 bits) and Source Sequence Count (14 bits).

(a) Segmentation Flags (Bits 0,1)

The Segmentation Flags occupy the two most significant bits of the 16-bit field. In the Source Packet, the Segmentation Flags shall always be set to "all ones".

(b) Source Sequence Count (Bits 2 through 15)

This 14-bit field contains a straight sequential count (module 16 384) of each packet generated by each unique source application process (as specified by the Application Process ID) on the spacecraft. The field will allow the ground telemetry acquisition systems to control the continuity of packet delivery for each Application Process ID.

During the continuous operation of a source application process, it is not permissible for the source to "short cycle" the sequence counter by resetting before the full counter accumulation has been reached; however, if the operation of a source is interrupted (e.g. through the power supply's being switched off), the source may start a new sequence count when its operation is resumed.

The source application process responsible for generating the Idle Source Packets (Application Process ID "all ones") is not required to maintain a Source Sequence Count.

B.5.4.2.1.3 Packet Length (16 Bits)

The Packet Length is a 16-bit field which specifies the number of octets contained within the Packet Data Field. The number is a binary value "C" expressed as follows:

C = [(Number of octets in Data Field) - 1]

Therefore, it should be noted that tne actual length of the entire Source Packet will implicitly be 6 octets longer, since the standard 48-bit Packet Header always precedes the Packet Datafield. Also, the smallest possible Source Packet length is 7 octets and the largest possible is 65 542 octets.

B 5.4.2.2 Packet Data Field

B 5.4.2.2.1 General

The Packet Data Field contains the information which is specific to the generating source on board the spacecraft. The only formal restriction imposed on this field is that its total length must be an integral number of octets; for the rest, the user will normally have complete freedom to specify the data content and the internal format of this field. However, users are coutioned that if the packet contents are to be processed within CCSDS agency support facilities, then local standards for internal formatting may be imposed. As an introduction to such standards, an optional subdivision of the Packet Data Field has been established, which is defined briefly in the following subsections. The detailed specification of such data structures and their protocols is not the subject of this Standard.

The subfields of the Packet Data Field are:

- Data Field Header (optional and variable)
- Source Data (variable)
- Packet Error Control (optional and variable)

B.5.4.2.2.2 Data Field Header

The Data field Header is an optional subdivision of the Packet Data Field.

The purpose of the Data Field Header is to provide a standard means for inserting within the first octets of a Source Packet Data field any ancillary data (time, additional packet type identification, internal data field format identification, etc.) which may be necessary to permit the interpretation of the source data contained within the packet by common data-processing facilities.

The presence or absence of a Data Field Header must be signalled by the Data Field Header Flag in the Packet Header.

The length of the Data Field Header shall be a multiple (integer) of octets. Comprehensive standards for the format of the Data Field Header are not the subject of this Standard.

B.5.4.2.2.3 Source Data

The Source Data field is a subdivision of the Packet Data Field. It is the user data in the form of a sequence of a sequence of octets.

B.5.4.2.2.4 Packet Error Control

The Packet Error Control field is an optional subdivision of the Packet Data Field.

The standard mechanisms that are used by the ground telemetry acquisition systems to extract Telemetry Packets from the Transfer Frame are such that no use is made of a Packet Error Control field (Telemetry Packets do not have such a field in any case). However, at the discretion of the user, an optional error detection code may be appended to the Source Data Field so that the ultimate recipient of the data is able to verify that the integrity of the complete Source Packet structure has been preserved during the entire transport process. The length of the Packet Error Control field shall be a multiple (integer) of octets. The coding and the format of the Packet Error Control field are not the subject of this Standard.

B.6 ELECTRICAL INTERFACES

B.6.1 Electrical Interfaces Overview

The overall PLM electrical blockdiagram is repeated in figure B.6.1-1. The various types of electrical interfaces provided by the PLM for EMI, EDI, and A.O. Instruments are shown in table B.6.1-1 and depicted in figure B.6.1-2. The shown values are preliminary instrument specific and defined in detail in the Instrument Specific ICD (ISAICD).

Electrical Interface	Number of Interfaces						
	IASi	MHS	ASCAT	MIMR	GOME	SCARAB	*AATSR*
Equipment Power	N & R	N & R		N & R	N	N & R	*N*
Equipment Heater Power	N & R	N & R		N & R	N	N & R	*N*
Heater Power for ICU	N & R	N & R		N & R		N & R	*N*
ICU Power (incl. DBU)	N & R	N & R	N & R	N & R	ScaRaB	N & R	*N*
Equipment SOL	N & R	N & R	N & R	N & R	N	N & R	*N*
Thermistor for Equipment	N & R	N & R		N & R	N	N & R	*N*
Thermistor for ICU	N & R	N & R		N & R	N	N & R	*N*
Depointing Signal Line	TBD	N & R		TBD		N & R	*N*
Deployment - Power (TBC)			N & R	TBD			*N 2)*
Depl.-Relay Status Acq. (TBC)			N & R	TBD			*N*
Depl.-Analog Acq (TBC)			N & R	TBD			
DBU Relay ON	N & R	N & R	N & R	N & R	ScaRaB	N & R	*N & R*
DBU Relay OFF	N & R	N & R	N & R	N & R	ScaRaB	N & R	*N & R*
Pyro Lines			N & R	TBD			
Instrument to Instrument I/F		8 Hz 1)	TBD	TBD			*TBD*
OBDH-BUS	N & R	N & R	N & R	N & R		N & R	*N & R*
Low Bit Rate Data I/F	N & R	N & R	N & R	N & R	N	N & R	*N & R*

Table B.6.1-1: Available/Used Electrical Interfaces

1) = 8 Hz from NIU for synchronization (TBC)
2) = AATSR uses auxiliary power during launch for locking the steering cooler

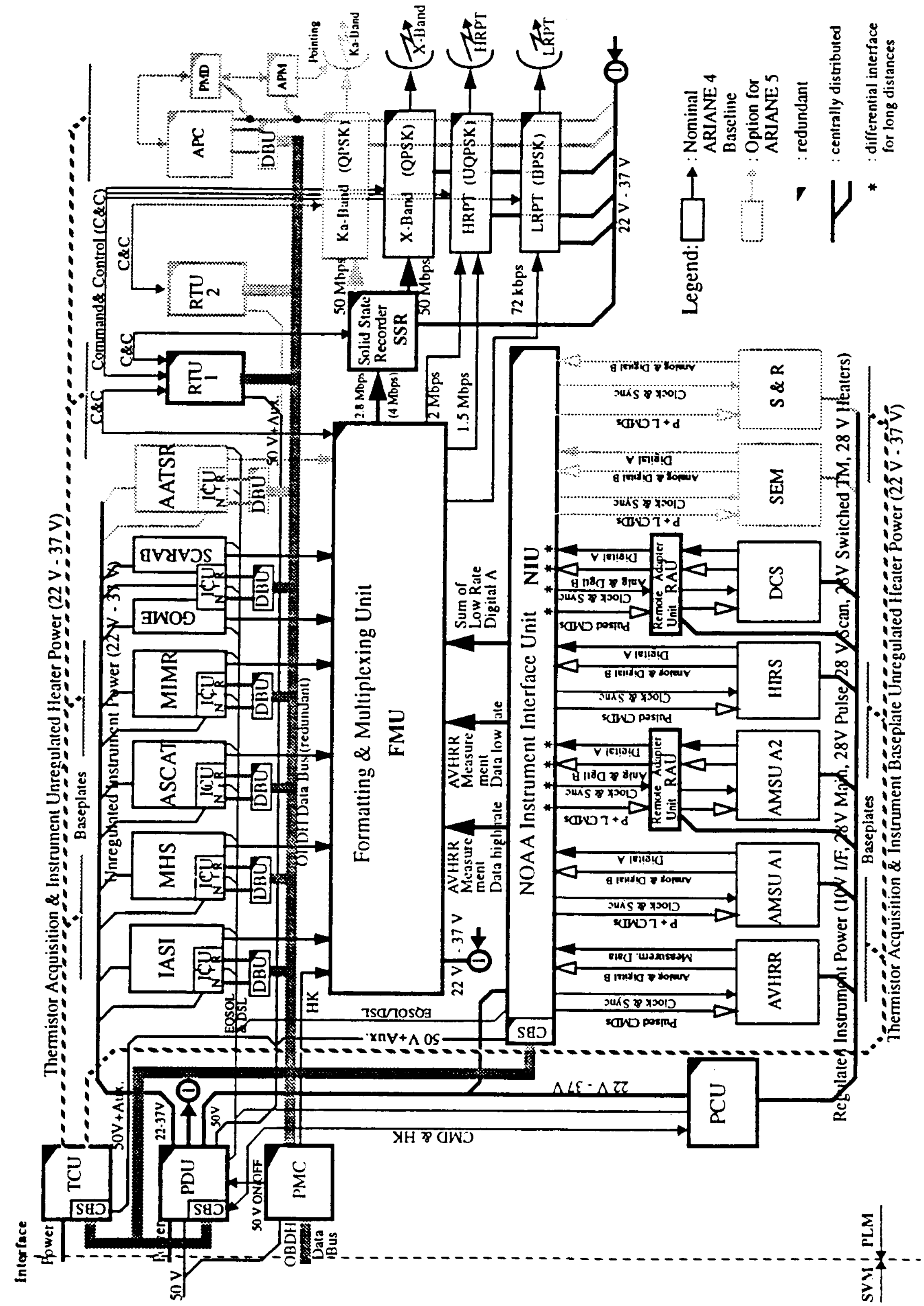


Figure B.6.1-1: Overall PLM Electrical Blockdiagram

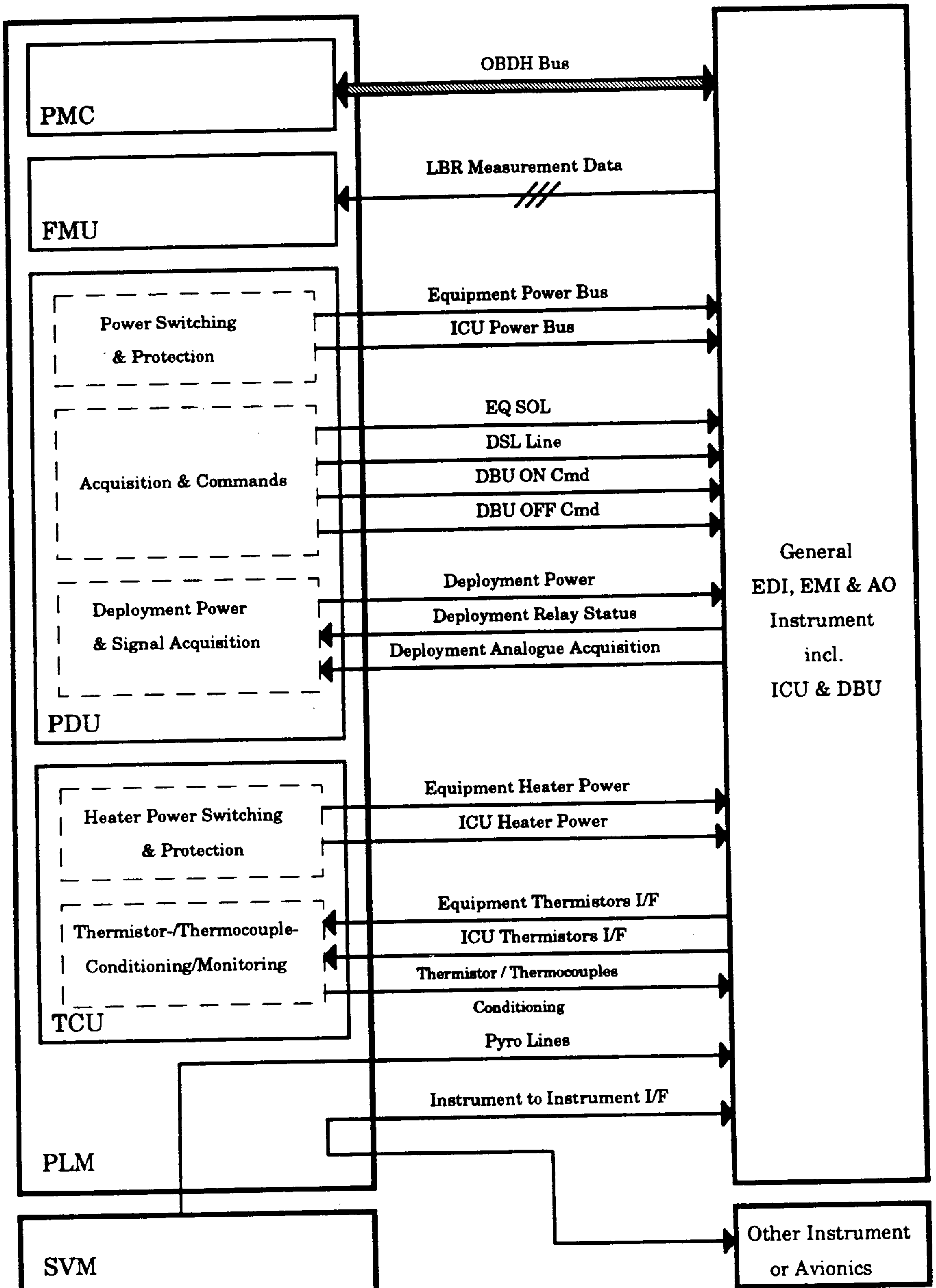


Figure B.6.1-2: Electrical Interfaces of EMI, EDI, and A.O. Instruments

B.6.2 Power Interfaces

B.6.2.1 Power Demand

The individual instrument power demands for BOL & EOL, and the required outlet dimensions from the PLM are defined in the Instrument Specific ICD.

EMI, EDI, and AO Instruments		Power in W, nominal Mode with recording	remark
EMI	MHS	100	
	IASI	216	
EDI	ASCAT	286	
	MIMR	171	
AO	SCARAB	75	
	GOME	44	
EMI, EDI, & AO total		892	

Table B.6.2.1-1: Preliminary Power Consumption Overview

B.6.2.2 Equipment Power Bus

B.6.2.2.1 Electrical Inteface Descrition

B.6.2.2.1.1. PLM Side

The instrument will get its equipment power via a unregulated commonly used equipment power bus in the voltage range between 22 to 37 V at the instrument connector bracket. The equipment power bus I/F circuit is as shown in figure B.6.2.2.1-1.
The Equipment Power line is protected against short circuit by one common current limiter (TBD) and switchable by one common switch on the PLM side. Except for fault conditions or spacecraft emergency, the PLM switch shall not be used to switch the connected instruments. The PLM provides redundant Equipment Power sources. Instruments having redundant circuits powered by the equipment power shall provide redundant equipment power connections. Instruments that do not have redundant circuits powered by the equipment power have a single equipment power connection.
Only one of the redundant equipment power bus lines is powered at a time (cold redundancy). Both power line returns (nominal and redundant) are connected to the electrical ground on the spacecraft side.
The PLM will provide the instrument with an equipment switch-off line (EQ SOL) for the redundant equipment power bus.
The various equipment power outlets are protected by fuses which limit the current as shown in table B.6.2.2.1-1 below:

Outlet w	Type of Fuser/Limiter	Maximum Specified Current
TBD	TBD	TBD
TBD	TBD	TBD
TBD	TBD	TBD

Table B.6.2.2.1-1: Equipment Power Outlet Current Limits

B.6.2.2.1.2. Instrument Side

Each instrument has to provide switchable power inputs and has to limit its current to a value of TBD times its nominal max. current for all instrument failure cases.
See description in the ISAICD.

B.6.2.2.2 Connectors

B.6.2.2.2.1. PLM Side

Table B.6.2.2.2-1 identifies the connector type used for the Equipment Power bus line (Low/Medium/High Power Bus) on the PLM side.

	Connector ID
nominal	P33 (low)
redundant	P34 (low)
nominal	P23 (medium)
redundant	P24 (medium)
nominal	P13 (high)
redundant	P14 (high)
Function	Equipment Power line from
	<low/medium/high> power bus
Type	
nominal	DBAS-76G-37-0-SW (low)
redundant	DBAS-76G-37-0-SX (low)
nominal	DBAS-76G-37-0-SW (medium)
redundant	DBAS-76G-37-0-SX (medium)
nominal	DBAS-76G-37-3-SW (high)
redundant	DBAS-76G-37-3-SX (high)

Table B.6.2.2.2-1: Connector Type PLM Side (TBC)

B.6.2.2.2.2. Instrument Side

Table B.6.2.2.2-2 identifies the connector type used for the Equipment Power bus line (Low/Medium/High Power Bus) on the instrument-side of each instrument.

	Connector ID
nominal	J33 (low)
redundant	J34 (low)
nominal	J23 (medium)
redundant	J24 (medium)
nominal	J13 (high)
redundant	J14 (high)
Function	Equipment Power line from <low/medium/high> power bus
	Type
nominal	DBAS-70-37-0-PW (low)
redundant	DBAS-70-37-0-PX (low)
nominal	DBAS-70-37-0-PW (medium)
redundant	DBAS-70-37-0-PX (medium)
nominal	DBAS-70-37-3-PW (high)
redundant	DBAS-70-37-3-PX (high)

Table B.6..2.2.2-2: Connector Type Instrument Side

T B D

Figure B.6.2.2.1-1: Circuit Diagram - Equipment Power (PLM Side)

B.6.2.2.3 Pin Allocation

Tables B.6.2.2.3-1/2/3 show the pin allocation of the Equipment Power bus (identical for the instrument and PLM side, as well as for the nominal and redundant connectors).

PIN No.	Function	Comment
1	BNR-CU-HPINST (+)	Twisted with 6
6	BNR-CU-HPINST (0V)	Twisted with 1
2	BNR-CU-HPINST (+)	Twisted with 9
9	BNR-CU-HPINST (0V)	Twisted with 2
3	BNR-CU-HPINST (+)	Twisted with 12
12	BNR-CU-HPINST (0V)	Twisted with 3
7	BNR-CU-HPINST (+)	Twisted with 8
8	BNR-CU-HPINST (0V)	Twisted with 7
10	BNR-CU-HPINST (+)	Twisted with 11
11	BNR-CU-HPINST (0V)	Twisted with 10
4	BNR-CU-HPINST (+)	Twisted with 5
5	BNR-CU-HPINST (0V)	Twisted with 4
Spare Pins	none	

Table B.6.2.2.3-1: Pin Allocation - High Power Bus (800 W) (P13, P14, J13, J14)

PIN No.	Function	Comment
2	BNR-CU-MPINST (+)	Twisted with 10
10	BNR-CU-MPINST (0V)	Twisted with 2
12	BNR-CU-MPINST (+)	Twisted with 3
3	BNR-CU-MPINST (0V)	Twisted with 12
22	BNR-CU-MPINST (+)	Twisted with 9
9	BNR-CU-MPINST (0V)	Twisted with 22
24	BNR-CU-MPINST (+)	Twisted with 23
23	BNR-CU-MPINST (0V)	Twisted with 24
11	BNR-CU-MPINST (+)	Twisted with 25
25	BNR-CU-MPINST (0V)	Twisted with 11
26	BNR-CU-MPINST (+)	Twisted with 27
27	BNR-CU-MPINST (0V)	Twisted with 26
14	BNR-CU-MPINST (+)	Twisted with 4
4	BNR-CU-MPINST (0V)	Twisted with 14
13	BNR-CU-MPINST (+)	Twisted with 28
28	BNR-CU-MPINST (0V)	Twisted with 13
29	BNR-CU-MPINST (+)	Twisted with 30
30	BNR-CU-MPINST (0V)	Twisted with 29
Spare Pins	1,5 to 8, 15 to 21, 31 to 37	

Table B.6.2.2.3-2: Pin Allocation - Medium Power Bus
(400 W) (P23, P24, J23, J24)

PIN No.	Function	Comment
2	BNR-CU-LPINST (+)	Twisted with 10
10	BNR-CU-LPINST (0V)	Twisted with 2
12	BNR-CU-LPINST (+)	Twisted with 3
3	BNR-CU-LPINST (0V)	Twisted with 12
22	BNR-CU-LPINST (+)	Twisted with 9
9	BNR-CU-LPINST (0V)	Twisted with 22
24	BNR-CU-LPINST (+)	Twisted with 23
23	BNR-CU-LPINST (0V)	Twisted with 24
11	BNR-CU-LPINST (+)	Twisted with 25
25	BNR-CU-LPINST (0V)	Twisted with 11
26	BNR-CU-LPINST (+)	Twisted with 27
27	BNR-CU-LPINST (0V)	Twisted with 26
Spare Pins	1,4 to 8, 13 to 21, 28 to 37	

Table B.6.2.2.3-3: Pin Allocation - Lower Power Bus (200 W) (P33, P34, J33, J34)

B.6.2.3 ICU Power Bus

B.6.2.3.1 Electrical Interface Description

B.6.2.3.1.1. PLM Side

The instrument will get its ICU power via the unregulated ICU power bus in the voltage range between 22 to 37 V at the instrument connector bracket. The ICU power bus I/F circuit is as shown in figure B.6.2.3.1-1.
the ICU Power is not used as heating power for the ICU; separate ICU heating power is available at the Heater Power bus connector (see chapter 2.2.2.3).
The ICU power line is protected against short circuit and individually switched on the PLM side.
The PLM provides redundant ICU Power only for instruments having redundant ICU´s. They shall then provide redundant ICU Power connection. Instruments that do not have a redundant ICU shall have a single ICU Power connection utilizing a single ICU Power line.
The PLM provides ICU power only on one of the redundant ICU power lines selecting thereby the ICU to be operated at a time. Both power line returns (nominal and redundant) are connected to the electrical ground on spacecraft side.
The ICU Power outlets are protected by fuses with a max. current as shown in table B.6.2.3.1-1 below:

Outlet Type	Maximum Specified Current
ICU	TBD

Table B.6.2.3.1-1: ICU Power Outlet Current Limits

B.6.2.3.1.2. Instrument Side

See description in the ISAICD.

T B D

Figure B.6.2.3.1-1: Circuit Diagram - ICU Power
(PLM Side)

B.6.2.3.2 Connectors

B.6.2.3.2.1. PLM Side

Table B.6.2.3.2-1 identifies the connector type used for ICU Power on the PLM side as well as EQ SOL, DSL and DBU Relay ON/OFF.

Connector ID	
nominal	P05
redundant	P06
Function	Control Signal I/F
Type	
nominal	KJL-6T-13-35 SN
redundant	KJL-6T-13-35 SC

Table B.6.2.3.2-1: Connector Type PLM Side

B.6.2.3.2.2. Instrument Side

Table B.6.2.3.2-2 identifies the connector type used forICU Power as well as EQ SOL, DSL and DBU Relay ON/OFF on the instrument side.

Connector ID	
nominal	J05
redundant	J06
Function	Control Signal I/F
Type	
nominal	KJL-3T-13-35 PN
redundant	KJL-3T-13-35 PC

Table B.6.2.3.2-2: Connector Type Instrument Side

B.6.2.3.3 Pin Allocation

Table B.6.2.3.3-1 shows the in allocation of the Control Signal interface (identical for the instrument and PLM side, as well as for the nominal and redundant connectors).

PIN No.	Function	Comment
21	SOL (1) (+)	
20	DBU Relay ON (+)	
19	DBU Relay OFF (+)	
18	DSL (+)	
17	ICU PWR (+)	
22	ICU PWR (+)	
13	ICU PWR (+)	
16	Common Return (0V)	
15	Common Return (0V)	
14	Common Return (0V)	
Spare Pins	1 through 12	
		Ground Shell is Ground

Table B.6.2.3.3-1: Pin Allocation Control Signal Interface

B.6.2.4 Heater Power

Only those instruments which require heater power have an interface to the Heater Power bus. Heater power can only be drawn in modes where the ICU does not perform thermal control.

B.6.2.4.1 Electrical Interface Description

B.6.2.4.1.1. PLM Side

The instruments which require heater power will get their heater power via unregulated Heater bus shown in figure B.6.2.4.1-1.
The heater power bus is individually protected against short circuit and individually switched on the PLM side.
The PLM provides redundant Heater Power sources. Instruments having redundant circuits powered by the Heater Power circuits shall provide redundant Heater Power connections. Instruments that do not have redundant circuits powered by the Heater Power circuits have a single Heater Power connection, in which case the spacecraft shall provide the necessary redundant source switching, utilizing a single Heater Power line.
Both lines (nominal and redundant) are powered simultaneously depending on which instrument thermostat circuit is active.
Power line return (nominal and redundant) are connected to the electrical ground on the spacecraft side.
The various heater power outlets are protected by fuses which limit the current as shown in table B.6.2.4.1-1 below:

Outlet W	Type of Fuse/Limiter	Maximum Specified Current
175	TBD	TBD
100	TBD	TBD
50	TBD	TBD
20	TBD	TBD

Table B.6.2.4.1-1: Heater Power Outlet Current Limits

B.6.2.4.1.2. Instrument Side

See description in the ISAICD.

TBD

Figure B.6.2.4.1-1: Circuit Diagram - Heater Power
(PLM Side)

B.6.2.4.2 Connectors

B.6.2.4.2.1 PLM Side

Table B.6.2.4.2-1 identifies the connector type used for the Heater Power line on the PLM side.

	Connector ID
nominal	P11
redundant	P12
Function	Heater Power
Type	
nominal	DBAS-76G-37-0-SB
redundant	DBAS-76G-37-0-SC

Table B.6.2.4.2-1: Connector Type PLM Side

B.6.2.4.2.2. Instrument Side

Table B.6.2.4.2-2 identifies the connector type used for the Heater Power line on the instrument side.

	Connector ID
nominal	J11
redundant	J12
Function	Heater Power
Type	
nominal	DBAS-70-37-0-PB
redundant	DBAS-70-37-0-PC

Table B.6.2.4.2-2: Connector Type Instrument Side

B.6.2.4.3 **Pin Allocation**

Table B.6.2.4.3-1 shows the pin allocation of the Heater Power line connectors (identical for the instrument and PLM side, as well as for the nominal and redundant connectors). Required pin numbers for the deployment bus are instrument specific and tbd. See instrument specific ICD.

PIN No.	Function	Comment
37	INST-HTR-PWR (+)	Twisted with 36
36	INST-HTR-PWR (0V)	Twisted with 37
35	INST-HTR-PWR (+)	Twisted with 34
34	INST-HTR-PWR (0V)	Twisted with 35
33	INST-HTR-PWR (+)	Twisted with 32
32	INST-HTR-PWR (0V)	Twisted with 33
31	ICU-HTR-PWR (+)	Twisted with 30
30	ICU-HTR-PWR (0V)	Twisted with 31
29	INST-DEPL-PWR (+)	Twisted with 28
28	INST-DEPL-PWR (0V)	Twisted with 29
27	INST-DEPL-PWR (+)	Twisted with 26
26	INST-DEPL-PWR (0V)	Twisted with 27
25	INST-DEPL-PWR (+)	Twisted with 24
24	INST-DEPL-PWR (0V)	Twisted with 25
23	INST-DEPL-PWR (+)	Twisted with 22
22	INST-DEPL-PWR (0V)	Twisted with 23
21	INST-DEPL-PWR (+)	Twisted with 20
20	INST-DEPL-PWR (0V)	Twisted with 21
PIN No.	Function	Comment
19	INST-DEPL-PWR (+)	Twisted with 18
18	INST-DEPL-PWR (0V)	Twisted with 19
17	INST-DEPL-PWR (+)	Twisted with 16
16	INST-DEPL-PWR (0V)	Twisted with 17
15	INST-DEPL-PWR (+)	Twisted with 14

14	INST-DEPL-PWR (0V)	Twisted with 15
12	INST-DEPL-PWR (+)	Twisted with 11
11	INST-DEPL-PWR (0V)	Twisted with 12
10	INST-DEPL-PWR (+)	Twisted with 9
9	INST-DEPL-PWR (0V)	Twisted with 10
Spare Pins		1 through 8, 13
Ground	Shell is Ground	

Table B.6.2.4.3-1: Pin Allocation Heater Power (TBC)

B.6.2.5 Deployment Power

The deployment power is allocated for non-pyro instrument deployment functions and fused and switched individually on PLM side. Details are tbc.
The power will be

B.6.2.5.1 Electrical Interface Description

B.6.2.5.1.1. PLM Side

The PLM provides redundant Deployment Power sources. Instruments having redundant Deployment Power circuits shall provide redundant Heater Power connections. Instruments that do not have redundant circuits have a single Deployment Power connection, in which case the spacecraft shall provide the necessary redundant source switching, utilising a single Deployment Power line. Power Line returns are connected to the electrical ground on spacecraft side,. Details are defined in the ISAICD.

B.6.2.5.1.2. Instrument Side

See description in the ISAICD.

B.6.2.5.2 Connectors

Auxiliary Power is available at the Heater Power line connector (see chapter 8.2.3). (tbc)

B.6.2.5.2.1. PLM Side

Table B.6.2.4.2-1 in the chapter above identifies the connector type used for the Depolyment Power line on the PLM side.
The PLM module provides red. Depl. Power .....
Depl. Power ..... (also when lanciers) can be individually swithced by the PLM.

B.6.2.5.2.2. Instrument Side

Table B.6.2.4.2-2 in the chapter above identifies the connector type used for the Deployment Power line on the instrument side.

B.6.2.5.3 Pin Allocation

Table B.6.2.4.3-1 in the chapter above shows the pin allocation of the Deployment Power line connectors (identical for the instrument and PLM side, as well as for the nominal and redundant connectors).

B.6.3 Command and Control Interface

B.6.3.1 OBDH Bus Interface

B.6.3.1.1 Electrical Interface Description

The communication link between the instrument ICU and the Payload Module Controller (PMC) uses a redundant OBDH Bus for the transfer of commands to the instrument and telemetry data from the instrument.

Through the use of the OBDH Bus, the PMC is able to transfer platform data and commands to the instrument ICU and to collect data, status, and housekeeping information from the instrument ICU.

The physical and low level protocol interface between the ICU and the OBDH Bus is via a Digital Bus Unit (DBU).

Commands from the PMC are transferred over the OBDH interrogation bus (I/B) to the ICU via the DBU. Telemetry data is returned to the PMC by the ICU over the response bus (R/B) via the DBU.

The OBDH interrogation bus is continuously driven by the PMC. Each interrogation bit is represented by Litton bit codes as depicted in figure B.6.3.1.1-1. Figure B.6.3.1.1-2 shows the Litton Code waveform.

Cable to be used: shielded twisted-pair cable, type GORE GSC-6509 Rev. 4 (GSC 7101 for testing purpose); the length between the instrument connector bracket and the DBU is defined in the ISAICD, chapter 6.5.

Voltage Level Litton (+)	up to +4.0 V
Voltage Level Litton (-)	up to -4.0 V
I/B Bus Receiver Impedance	> 10 k < 100 pF (300 kHz to 2 MHz)

Table B.6.3.1.1-1: Litton Characteristics Receiver

Zero Voltage Level	-0.2 Vp to +0.2 Vp
Voltage Level Litton (+)	+2.8 V to +4.0 V
Voltage Level Litton (-)	-2.8 V to -4.0 V
Duty Cycle	tp/T = 0.5 25 % instrument side tp/T = 0.6 5 % PMC/EGSE side
Rise Time	100 ns max.
Fall Time	100 ns max.
Source Impedance in active status	< 30
Source Impedance in inactive status	> 150 pF (300 kHz to 2 MHz)

Table B.6.3.1.1-2: Litton Characteristics Transmitter

Notes:

1. Rise and fall times are measured between levels corresponding to 10 % and 90 % of BA for positive pulse or of BC for negative pulse.
2. Pulse width and period are emeasured at level corresponding to 50 % of BA or of BC.
3. The overshoot AD, BD, and CD shall be ot more than 10 % of the full amplitude of the corresponding pulse (BA or BC).

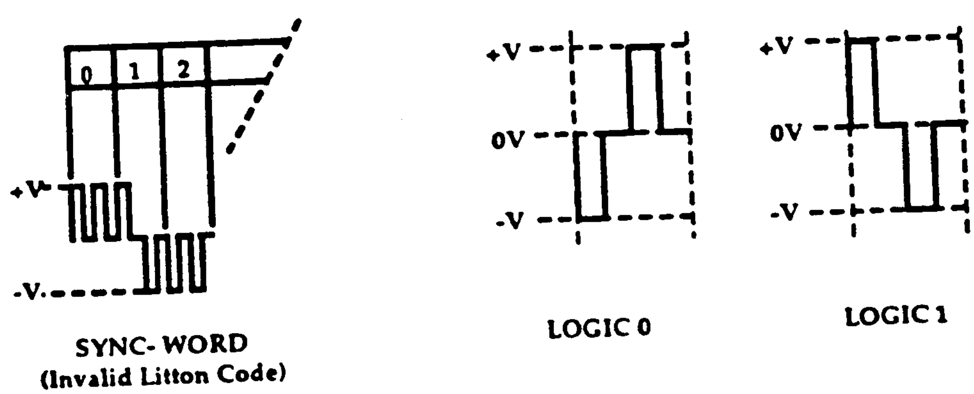


Figure B.6.3.1.1-1: Litton Code

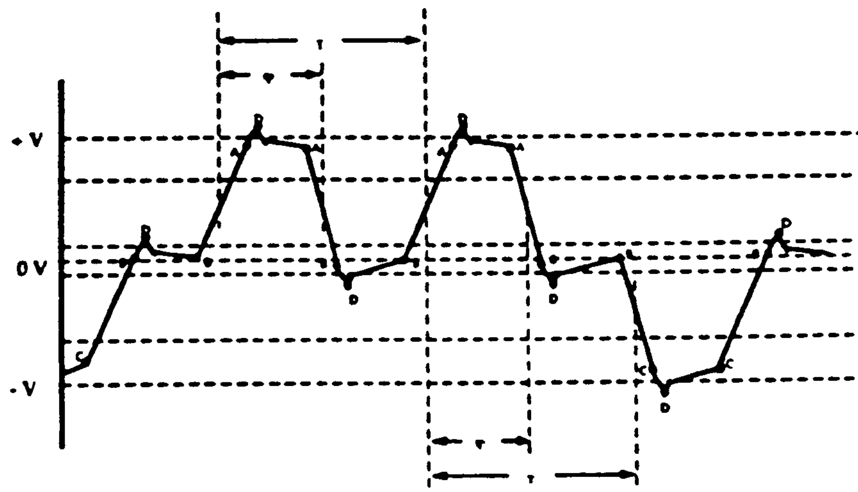


Figure B.6.3.1.1-2: OBDH Busses Waveform

B.6.3.1.2 Connectors

The DBU is equipped with four connectors dedicated to the payload OBDH Bus interface.

With respect to this, six connectors (see figures B.6.3.1.2-1/2) are described, due to the fact that the DBU is a Customer Furnished Equipment which is integrated into the instrument.

The connector types and pin allocations for the interfaces between PLM, the OBDH-Bus and the DBU are shown in the following tables.

B.6.3.1.2.1 Connector Type, OBDH Bus at the PLM I/F

Tables B.6.3.1.2.1-1/2 identify the connector types used for the OBDH bus line at the interface to the PLM.

Connector ID nominal redundant	 P01 P02
Function	OBDH-bus
Type nominal redundant	 KJL-6T-15-35 SB KJL-6T-15-35 SA

Table B.6.3.1.2.1-1: Connector Type PLM side

Connector ID nominal redundant	 J01 J02
Function	OBDH-bus
Type nominal redundant	 KJL-3T-15-35 PB KJL-3T-15-35 PA

Table B.6.3.1.2.1-2: Connector Type Instrument side

B.6.3.1.2.2 Pin Allocation

Table B.6.3.1.2.2-1 shows the pin allocation of the OBDH bus interface to the PLM (identical for the instrument and PLM side as well as for the nominal and redundant connectors).

PIN No.	Function	Comment
37	Interrogation Bus (S) IN	Twisted with 36
36	Interrogation Bus (R) IN	Twisted with 37
35	Screen for 36, 37	
21	Interrogation Bus (S) OUT	Twisted with 22
22	Interrogation Bus (R) OUT	Twisted with 21
32	Screen for 21, 22	
23	Response Bus (S) IN	
24	Response Bus (R) IN	Twisted with 23
33	Screen for 23, 24	
25	Response Bus (S) OUT	Twisted with 26
26	Response Bus (R) OUT	Twisted with 25
34	Screen for 25, 26	
Spare Pins		1 through 20, 27 through 31

Table B.6.3.1.2.2-1: Pin Allocation OBDH Bus (I/F to the PLM)

Abbreviations:

(S) = Signal IN ...= to the instrument
(R) = Return OUT = from the instrument

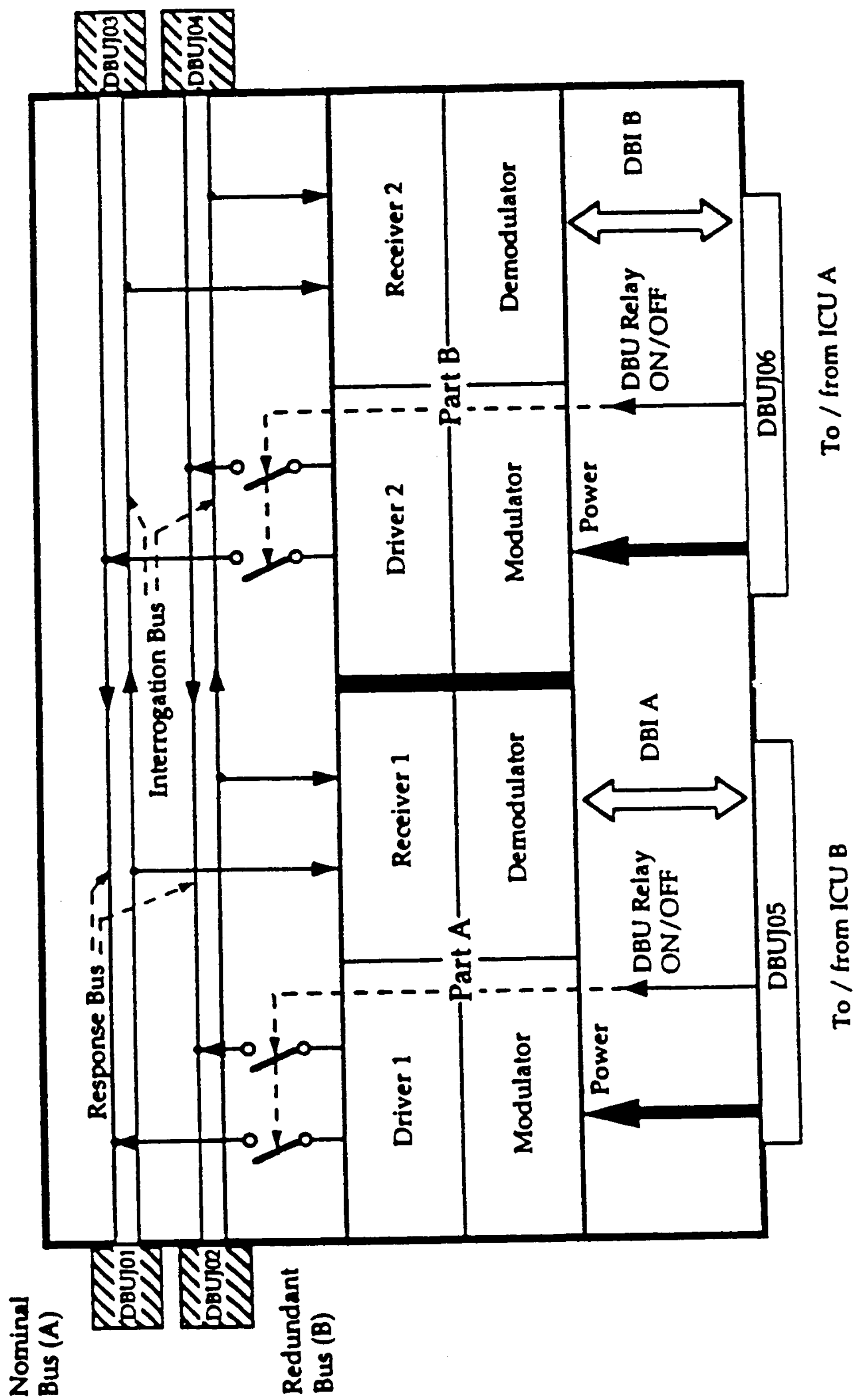


Figure B.6.3.1.2.2-1: DBU Connector Assignment

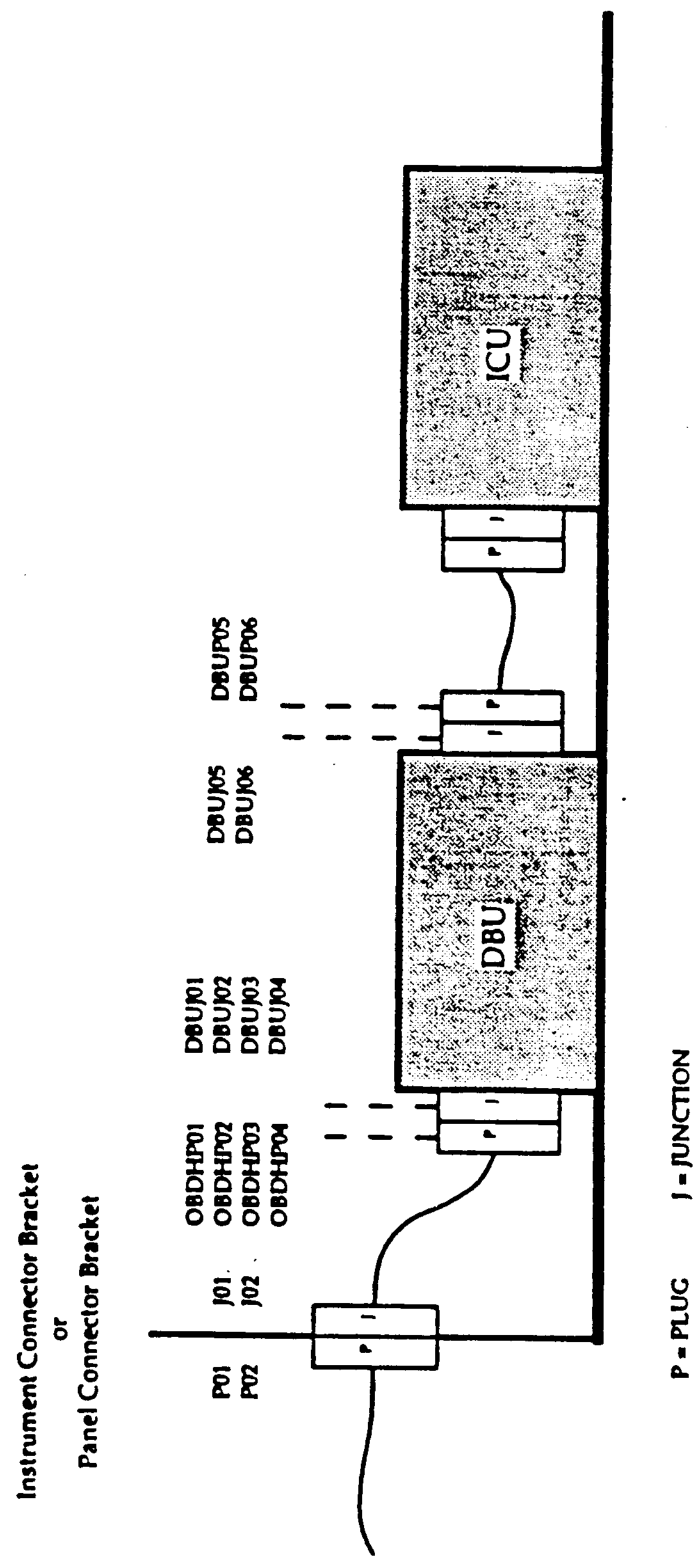


Figure B.6.3.1.2.2-2: DBU/ICU/Connector Bracket - Connector Assignment

B.6.3.1.3.3 Connector Type, OBDH Bus at the DBU

Tables B.6.3.1.2.3-1 to B.6.3.1.2.3-8 identify the connector types used for the OBDH bus line at the interface to the DBU.

Connector ID	OBDHP02
Function	OBDH-Bus IN, nominal
Type	DEM9SNMBOL3

Table B.6.3.1.2.3-1: Connector Type, Harness

Connector ID	DBUJ02
Function	OBDH-Bus IN, nominal
Type	DEM9PNMBOL3

Table B.6.3.1.2.3-2: Connector Type, DBU

Connector ID	OBDHP01
Function	OBDH-Bus IN, redundant
Type	DEM9SNMBOL3

Table B.6.3.1.2.3-3: Connector Type, Harness

Connector ID	DBUJ01
Function	OBDH-Bus IN, redundant
Type	DEM9PNMBOL3

Table B.6.3.1.2.3-4: Connector Type, DBU

Connector ID	OBDHP04
Function	OBDH-Bus OUT, nominal
Type	DEM9PNMBOL3

Table B.6.3.1.2.3-5: Connector Type, Harness

Connector ID	OBUJ4
Function	OBDH-Bus OUT, nominal
Type	DEM9SNMBOL3

Table B.6.3.1.2.3-6: Connector Type, DBU

Connector ID	OBDHP03
Function	OBDH-Bus OUT redundant
Type	DEM9PNMBOL3

Table B.6.3.1.2.3-7: Connector Type, Harness

Connector ID	DBUJ03
Function	OBDH-Bus OUT, redun-dant
Type	DEM9SNMBOL3

Table B.6.3.1.2.3-8: Connector Type, DBU

B.6.3.1.3.4 Pin Allocation

Table B.6.3.1.2.4-1 shows the pin allocation of the OBDH-bus interface at the DBU (identical for the OBDHP01 to OBDHP04 and DBUJ01 to DBUJ04 connectors).

PIN No.	Function	Comment
1	Interrogation Bus (S)	
2	N.C.	
3	N.C.	
4	Response Bus (S)	
5	N.C.	
6	Interrogation Bus (R)	
7	Chassis Ground	
8	Response Bus (R)	
9	N.C.	

Table B.6.3.1.2.4-1: Pin Allocation

B.6.3.2 ICU to DBU Interface

B.6.3.2.1 Electrical Interface Description

The DBU is internally redundant, containing two separate DBI functions as shown in figure B.6.3.2.1-1.

Each redundant ICU has its own dedicated DBI and there is no cross-strapping permitted between ICU(A) and DBU(B), or between ICU(B) and DBU(A) at the DBI interface. When an ICU is powered up it shall automaically provide power to the corresponding part of the DBU. Each half DBU, when powered, automatically operates on whichever of the 2 redundant OBDH busses is active.

The signals which constitutde the DBI are shown in figure B.6.3.2.1-2.

Command messages are made up of a sequence of 32 bit interrogation words received from the OBDH Interrogation bus. The 32 bit words are passed to the ICU in serial form over the DBI.

The OBDH Interrogation Bus is continuously activc (except for failure cases), making the DBI continuously active when powered.

ICU responses are made up of sequencs of either 21-bit or 13-bit serial words passed from the ICU to the OBDH response bus via the DBI and DBU. The response channel remains inactive unless responding to specific interrogations.

The timing of interrogations and responses has a fixed relationsip giving a fully synchronized interface. A continuous clock (RIR-CLK) and broadcast pulses shall be used to synchronize operations.

To prevent that the ICU emits spurious signals on the response bus during "power on", the DBU response bus relay will be closed by the PLM with an appropriate delay of 500 mµsec in which the ICU shall reache a stable condition.

To prevent that spurious signals are emitted on the response bus during power down, the response bus relay in the DBU will be opened by the PLM before the ICU Power Bus is switched off with a delay corresponding to a time greater than the "Response Bus Relay" response time.

<u>The signal lines involved are</u>:

ICU Inputs from DBU:

RIR CLK
Interrogation data clock (524,288 Hz) is recovered from the OBDH Litton coded interrogations. The clock frequency can vary between 524,288 ± 5000 Hz.

RIR DATA
Interrogation data (32-bit interrogation words) in serial NRZ-L form, recovered from the Litton code.

RIR SYNC
Sync pulse used as a marker with respect to the 32-bit interrogation word, retried from the sync pattern in the Litton message.

RIR VAL
Valid interrogation flag set HI to indicate the interrogation is valid. HI to LO transition caused by either:

- receipt of an interrogation bit inorrectly Litton coded (Litton Error) causing a HIL to LO transition on the RIR-CLK falling edge following the bit in error

- loss of OBDH interrogation sync pattern within the expected time interval.

ICU Outputs to DBU:

RRT-EN
Response transmission enable signal to gate the ICU response word through to the OBDH response bus.

RRT-DATA
Response data words (21-bit or 13-bit) in serial NRZ-L form to be transferred to the OBDH response bus in Litton coded form.

<u>Timing</u>:

The general timing relationship between interrogations and corresponding responses is shown in figure B.6.3.2.1-3.

The relative timing of the DBI signals is shown in figure B.6.3.2.1-4.

The following clarifications apply to figure B.6.3.2.1-3 (overal timing) and to figure B.6.3.2.1-5 (details for RIR-VAL):

- All timing on DBI is referred to one reference, the RIR-CLK signal.
- The timing applies to the DBU-side of the interface. This means that for all handshake timing, twice the delay on the lines has to be considered.
- Note that between ICU input level detection and output level transmission a fixed time relationship must be achieved.

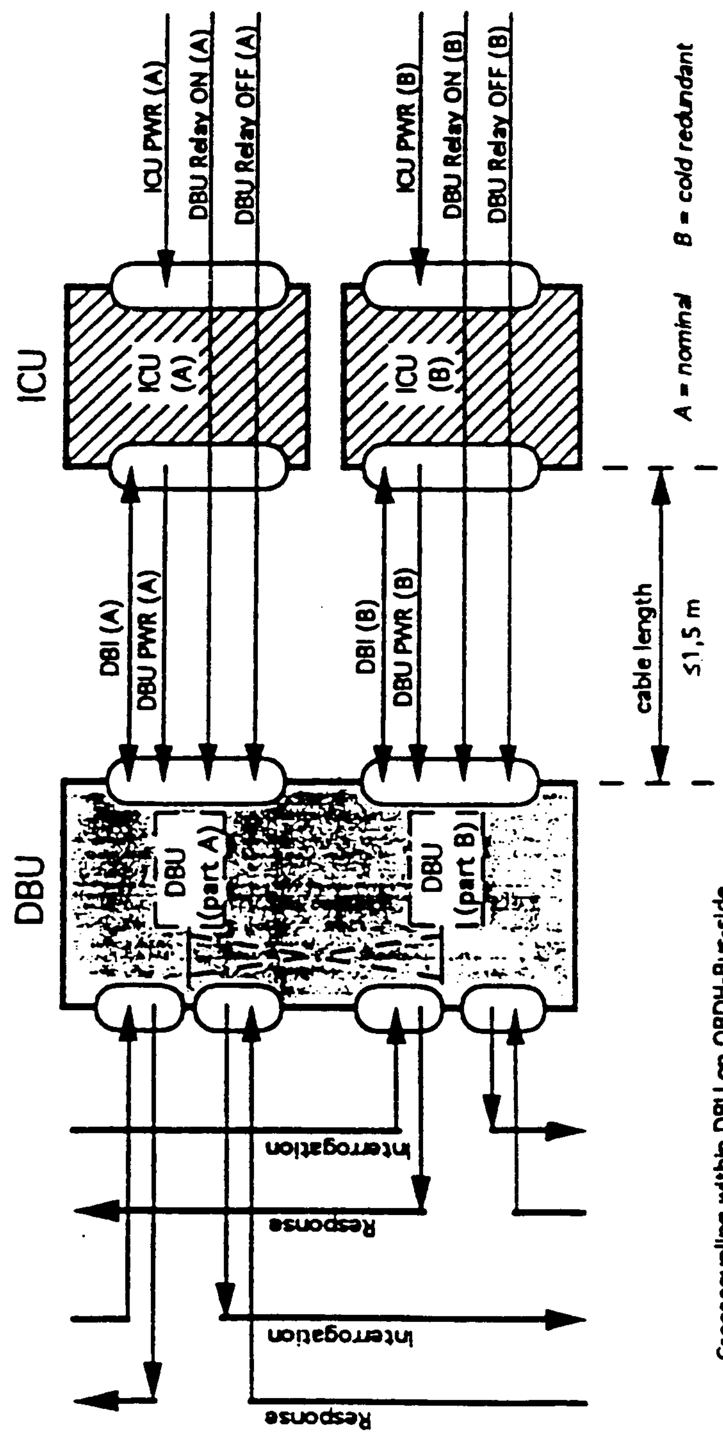


Figure B.6.3.2.1-1: DBU/ICU Interface

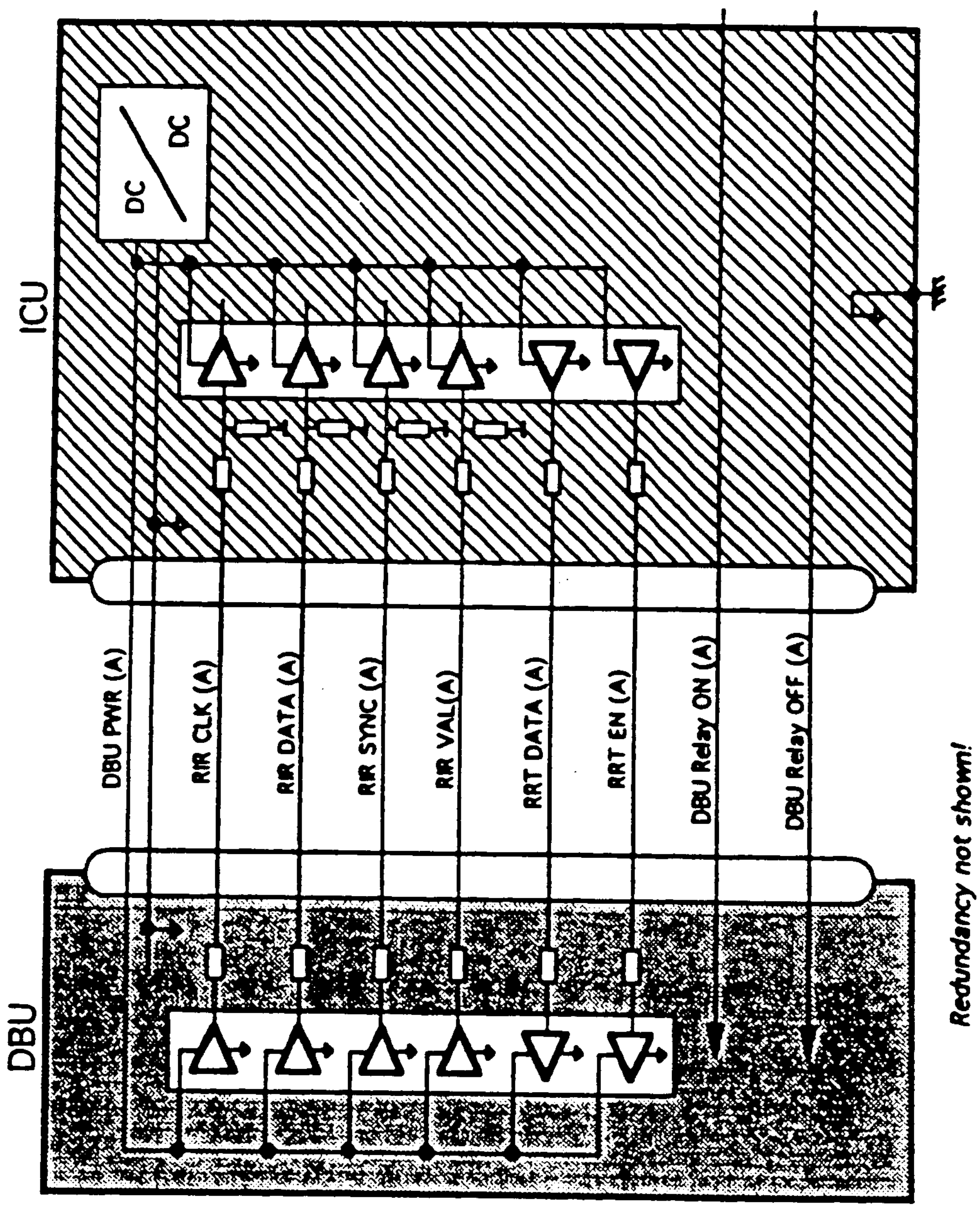


Figure B.6.3.2.1-2: Digital Bus Interface Lines

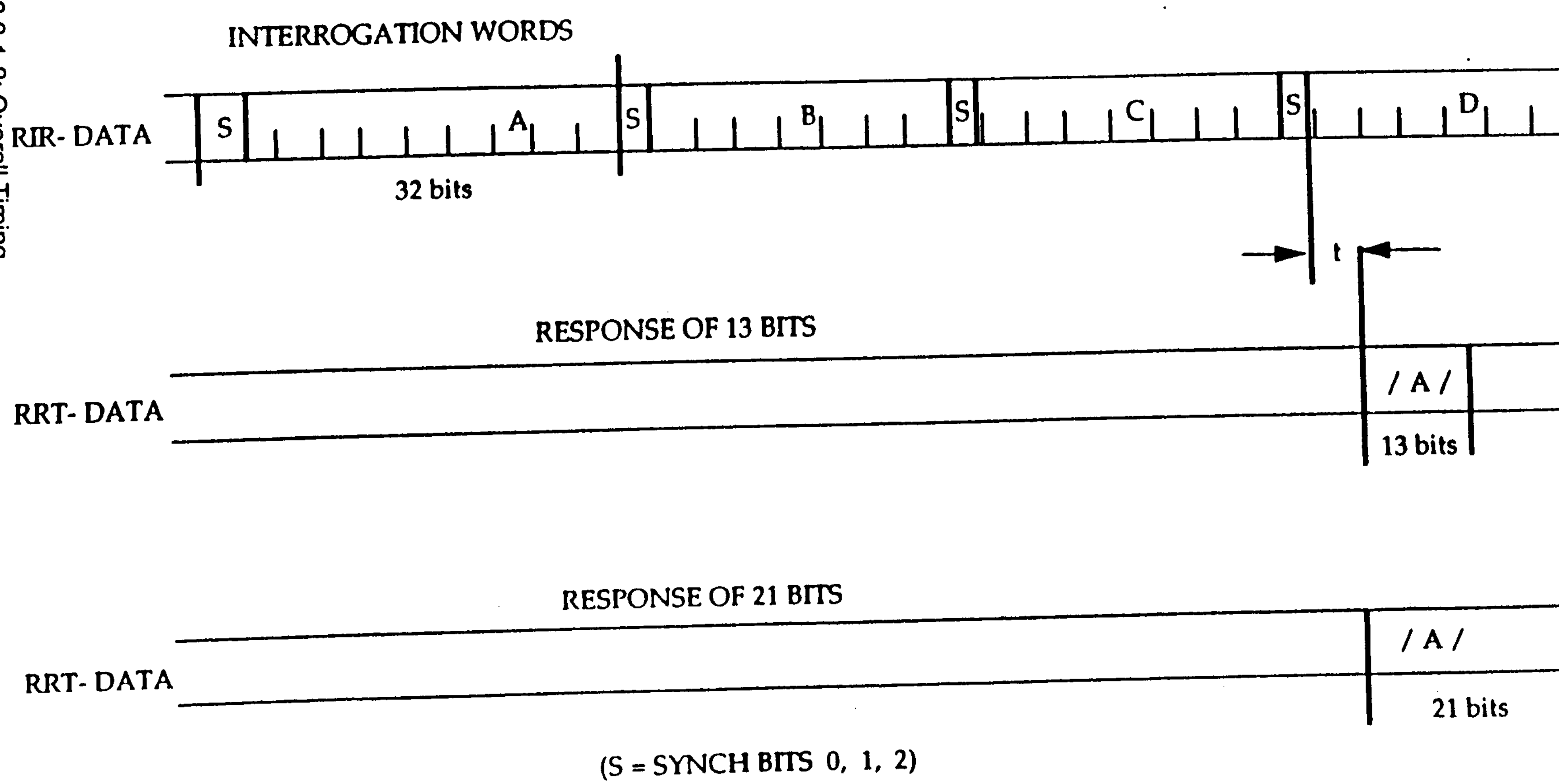
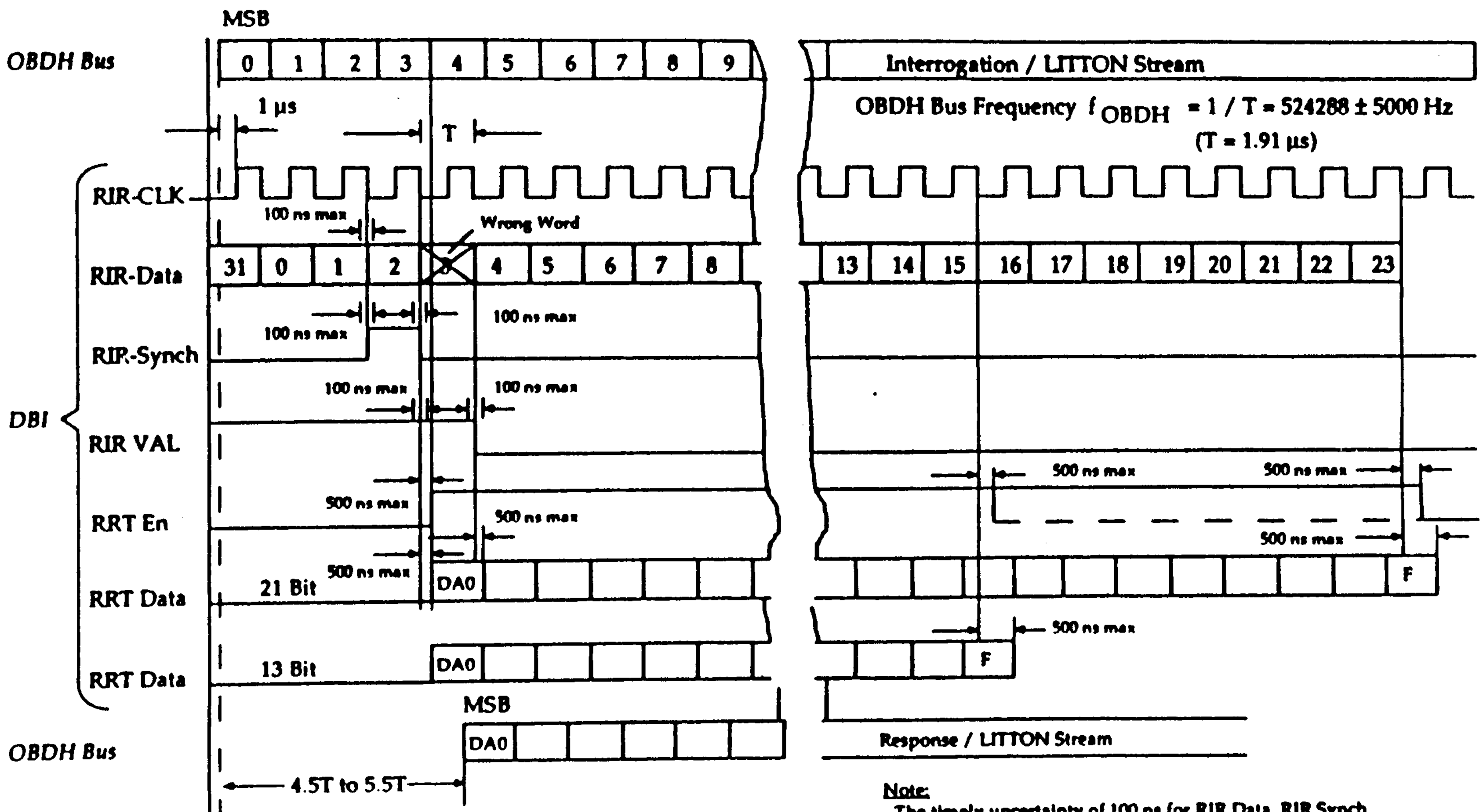
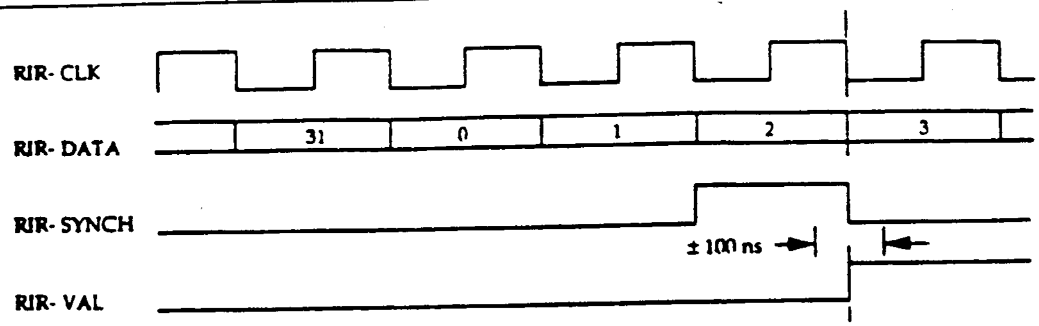


Figure B.6.3.2.1-3: Overall Timing

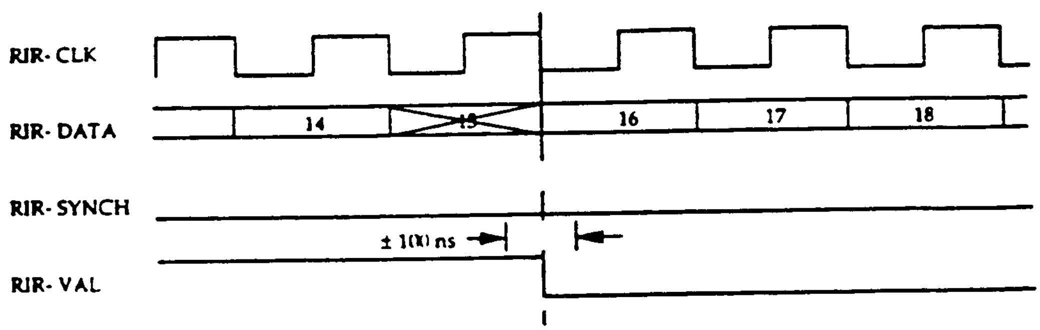


Note:
- The timely uncertainty of 100 ns for RIR Data, RIR Synch, and RIR VAL w.r.t falling edge of RIR CLK is to be understood "to each side"
- only true/ set state as up diagram shown
- all timing refers to DBU connector
- Synch Bits 0, 1, 2 are represented by logical "1", "1", "0"
- RIR CLK duty cycle (50 ± 5) %
- F: End of response Bit (fixed to zero)

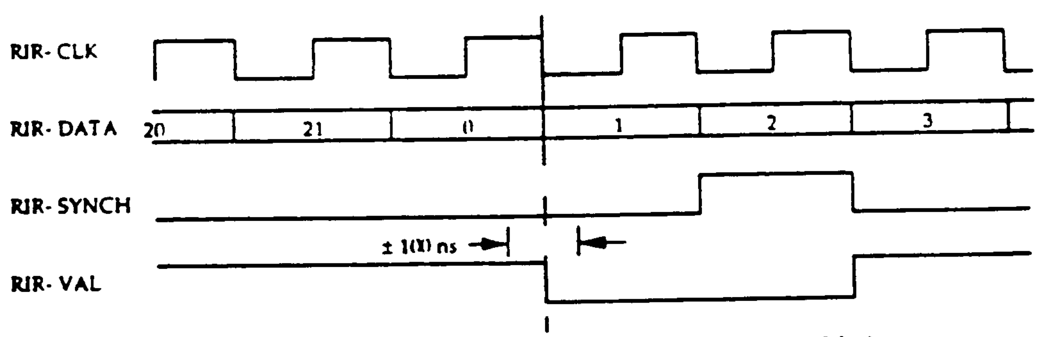
Figure B.6.3.2.1-4: DBI Timing



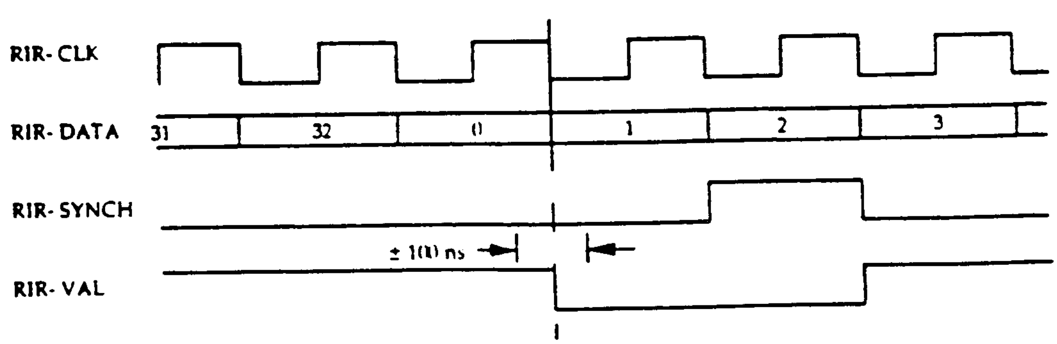
Timing Detail 1: Setting of RIR- VAL



Timing Detail 2: Resetting of RIR- VAL after a wrong bit



Timing Detail 3: Resetting of RIR- VAL (less than 29 bits)



Timing Detail 4: Resetting of RIR- VAL (more than 29 bits)

Figure B.6.3.2.1-5: Details for RIR-VAL Timing

Driver characteristics:

Encoding	NRZ-L
Drive Type	Single-ended CMOS
Output Voltage Levels: logical "zero" logical "one"	 V_{OL}: 0 V to +0.4 V V_{OH}: +4.5 V to +5.5 V
Resistor in Output Line	820 5 % (serial)
Overvoltage Capability	0 V to +7 V with 700 series resistor
Fault Condition	0 V to +7 V, 700
Circuit Implementation	54 HC 244

Table B.6.3.2.1-1: ICU/DBU Driver Characteristics

The given circuit characteristics shall be verified in accordance with the conditions specified in the data sheets.

Receiver characteristics:

Encoding	NRZ-L
Receive Type	Single-ended CMOS
Input Voltage Threshold: logical "zero" logical "one"	 V_{T-} +1.35 V V_{T+} + 4.2 V
Resistor in Input Line	1 k (serial) (TBC)
Over-vltage Capability	0 V to +7 V with 700 series resistor
Rise/Fall Time Capability	280 nS between 10 % and 90 % of signal
Circuit Implementation	54 HC 14

Table B.6.3.2.1-2: ICU/DBU Receiver Characteristics

The given circuit characteristics shall be verified in accordance with the conditions specified in the data sheets.

B.6.3.2.2 Connectors

Tables B.6.3.2.2-1/2 identify the connector types used for the DBI.

Connector ID nominal redundant	 DBUJ06 DBUJ05
Function	DBI
Type nominal redundant	 DBM25PNMBOL3 DBM25PNMBOL3

Table B.6.3.2.2-1: Connector Type from DBU

Connector ID nominal redundant	 DBIP06 DBIP05
Function	DBI
Type nominal redundant	 DBM25SNMBOL3 DBM25SNMBOL3

Table B.6.3.2.2-2: Connector Type to ICU

B.6.3.2.3 Pin Allocation

Table B.6.3.2.3-1 sows the pin allocation of the DBI at the DBU (identical for the DBU and ICU side, as well as for the nominal and redundant connectors).

PIN No.	Function	Comment
1	Power Return & Logic Ground	
2	Reserved	
3	RRT-EN	
4	RRT-Data	
5	RIR-VAL	
6	RIR-CLK	
7	RIR-Data	
8	RIR-SYNC	
12	Power Supply +6 V	
13	Reserved	
14	Reserved	
15	Reserved	
16	Reserved	
17	Reserved	
18	Reserved	
19	Reserved	
20	DBU Relay On	
21	DBU Relay On Return	
22	DBU Relay Off	
23	DBU Relay Off Return	
24	Chassis Ground	
25	Chassis Ground	
Spare Pins		9, 10, 11

Table B.6.3.2.3-1: Pin Allocation DBI

B.6.3.4 Control Signal Interfaces

B.6.3.4.1 Electrical Interface Description

B.6.3.4.1.1 PLM Side

The PLM supplies each instrument with hot redundant Equipment Switch Off Lines (EQ SOL), hot redundant Depointing Signal Lines (DSL) and cold redundant DBU Relay ON/OFF lines.

The connection between each redundant part of the DBU and the OBDH Response Bus is interruptable by latch relays.

Each relay is activated/deactivated upon reception of the relevant ON/OFF command, issued by the PLM.

The **DSL** and **EQ SOL** source characteristics are summarized in table B.6.3.4.1.1-1.

Bi-level pulse signal	
Quiescent state voltage	0 V to 1.5 V
Active state voltage	1B.6.3 V to 37 V
Pulse duration	45 ms to 150 ms
Bounce time	10 ms
Source impedance	2

Table B.6.3.4.1.1-1: DSL and EQ SOL Source Characteristics

The **DBU Relay ON/OFF** receiver characteristics are given in table B.6.3.4.1.1-2.

Signal levels	
Quiescent state voltage	0 V to 1.5 V
Active state voltage	11 V to 13 V
Duration	50 ms to 150 ms
Rise/Fall time	10 ms
Fault Voltage	16 V
Relay Coil Resistance	500 10 %
Current limitation	150 mA max.

Table B.6.3.4.1.1-2: DBU Relay Receiver Characteristics

B.6.3.4.1.2 Instrument Side

The Depointing Signal Lines (DSL) are routed to the instrument ICU and to equipment units if necessary.

The Equipment Switch Off Lines (EQ SOL) are routed to the instrument ICU and in parallel to the instrument centralized EQ PWR switch off functions.

The nominal control signals are connected to the nominal instrument units.

The redundant control signals are connected to the redundant instrument units.

The common return lines (see the pin allocation) are to be used. There is no connection to other return lines.

On occurrence of the DSL signal the instrument shall establish protective measures against non-nominal sun illumination.

On activation of the EQ SOL all instrument units, connected to the Equipment Power Bus (B.6.2.2) shall be switched off. This shall be achieved by an ICU independent function to ensure the switch-off even in the case of an ICU failure.

The instrument ICU shall receive the EQ SOL in form of an interrupt and shall adapt and continue its monitoring to the achieved mode after power down of all equipment.

The instrument shall route the DBU Relay ON/FF lines through the ICU to the DBU without any function within the ICU.

The load for the **DSL** and **EQ SOL** shall be within the envelope specified in table B.6.3.4.1.2-1.

Load current at 37 V	125 mA
Reverse voltage range on active to quiescent transition	0 V ... -1.5 V
The instrument loads shall be equipped with appropriate circuits in order to suppress any switching transients, in particular those due to inductive loads such as relays, which may cause the current driving capability or the overvoltage capability of the source to be exceeded.	

Table B.6.3.4.1.2-1: DSL and EQ SOL Load Characteristic

B.6.3.4.2 Connectors

B.6.3.4.2.1 PLM Side

Table B.6.3.4.2.1-1 identifies the connector type used for EQ SOL, DSL and DBU Relay ON/OFF (as wel as ICU Power, see chapter 8.2.2) on the PLM side.

Connector ID nominal redundant	 P05 P06
Function	Control Signal I/F
Type nominal redundant	 KJL-6T-13-35 SN KJL-6T-13-35 SC

Table B.6.3.4.2.1-1: Connector Type PLM side

B.6.3.4.2.2 Instrument Side

Table B.6.3.4.2.2-1 identifies the connector type used for EQ SOL, DSL and DBU Relay ON/OFF (as well as ICU Power, se chapter 8.2.2) on the instrument side.

Connector ID nominal redundant	 J05 J06
Function	Control Signal I/F
Type nominal redundant	 KJL-3T-13-35 PN KJL-3T-13-35 PC

Table B.6.3.4.2.2-1: Connector Type Instrument side

B.6.3.4.3 Pin Allocation

Table B.6.3.4.2.3-1 shows the pin allocation of the Control Signal interface (identical for the instrument and PLM side, as well as for the nominal and redundant connectors).

PIN No.	Function	Comment
21	SOL (1) (+)	
20	DBU Relay ON (+)	
19	DBU Relay OFF (+)	
18	DSL (+)	
17	ICU PWR (+)	
22	ICU PWR (+)	
13	ICU PWR (+)	
16	Common Return (0 V)	
15	Common Return (0 V)	
14	Common Return (0 V)	
Spare Pins		1 through 12
Ground	Shell is Ground	

Table B.6.3.4.2.3-1: Pin Allocation Control Signal Interface

B.6.3.5 Pyrotechnic Interfaces

B.6.3.5.1 Electrical Interface Description

B.6.3.5.1.1 PLM Side

Instruments requiring pyros for the deployment of antennas etc. are provided with pyro interfaces as required in the ISAICD. The satellite level distribution system incorporates an externally accessible safeplug installed on the Service Module which interrupts the pyrotechnic lines. The firing procedure and sequence is TBD.

The source characteristics are summarized in table B.6.3.5.1.1-1.

Firing Current	5.2 A
Open Circuit Voltage	17.7 V to 22 V
Active state voltage	1B.6.3 V to 37 V
Pulse duration	45 ms 10 ms
Static Discharge Resistor	100 k to 1 M
Source impedance	1.72
In the non-fire configuration, the firing relay will place a 47 resistor in parallel to the initiator.	

Table B.6.3.5.1.1-1: Source Characteristics

B.6.3.5.1.2 Instrument Side

The firing circuits are not routed in multi-pin connectors with other load carrying circuits, except other pyro circuits.

The pyrotechnic devices meet the electrical requirements as shown in table B.6.3.5.1.2-1.

Primary Device	Dassault 1 TAP WH40
No-fire current	1.0 A, 1 W for 5 min
Input resistance (bridgewire)	1.05 0.1
Safe no-fire current for testing	10 mA
Initiator functioning time	10 ms at 5.2 A

Table B.6.3.5.1.2-1: Interface Requirements

See further instrument specific description in the ISAICD.

B.6.3.5.2 Pyrotechnic Connectors

B.6.3.5.2.1 PLM Side

Table B.6.3.5.2.1-1 identifies the connector type used for the Pyrotechnic interface on the PLM side.

Connector ID	
nominal	P07
redundant	P08
Function	Pyrotechnic Interface
Type nominal redundant	 KJL-6T-13-35 SA KJL-6T-13-35 SD

Table B.6.3.5.2.1-1: Connector Type PLM side

B.6.3.5.2.2 Instrument Side

Table B.6.3.5.2.2-1 identifies the connector type used for the Pyrotechnic interface on the instrument side.

Connector ID	
nominal	P07
redundant	P08
Function	Pyrotechnic Interface
Type nominal redundant	 KJL-3T-13-35 PA KJL-3T-13-35 PD

Table B.6.3.5.2.2-1: Connector Type Instrument side

B.6.3.5.3 Pin Allocation

Table B.6.3.5.2.3-1 shows the pin allocation of the Pyrotechnic interface (identical for the instrument and PLM side, as well as for the nominal and redundant connectors).

PIN No.	Function	Comment
22	INSTR. (XX) PYRO 1 (S)	Twisted with 21
21	INSTR. (XX) PYRO 1 (R)	Twisted with 22
20	INSTR. (XX) PYRO 2 (S)	Twisted with 19
19	INSTR. (XX) PYRO 2 (R)	Twisted with 20
18	INSTR. (XX) PYRO 3 (S)	Twisted with 17
17	INSTR. (XX) PYRO 3 (R)	Twisted with 128
16	INSTR. (XX) PYRO 4 (S)	Twisted with 15
15	INSTR. (XX) PYRO 4 (R)	Twisted with 16
14	INSTR. (XX) PYRO 5 (S)	Twisted with 13
13	INSTR. (XX) PYRO 5 (R)	Twisted with 14
12	INSTR. (XX) PYRO 6 (S)	Twisted with 11
11	INSTR. (XX) PYRO 6 (R)	Twisted with 12
Spare Pins		1 through 10

Table B.6.3.5.2.3-1: Pin Allocation Pyrotechnic interface

<u>Abbreviations:</u>

(S) = Signal (R) = Return

Cable used: shielded twisted-pair AWG 20

B.6.3.6 Thermistor Interfaces

B.6.3.6.1 Electrical Interface Description

B.6.3.6.1.1 PLM Side

The PLM supplies heater power and provides thermistor monitoring for externally mounted elements when the instrument is in STBY or lower mode. The thermistor conditioning circuits will be on PLM side.

The TCU PLM interfaces to all thermistors in a manner which allows the PLM to have full access to all thermistors..

B.6.3.6.1.2 Instrument Side

For elements mounted outside the PLM Structure, the requirements for the thermistor I/F as shown in table B.6.3.6.1.2-1 shall apply.

Temperature range of thermistor	-60°C to +80°C
Thermistor impedance over temperature range -60°C to +80°C	3.74 k to 2.21 k
Zero power resistance at 25°C	15 k
Tolerance % R, range 0°C to +75°C	0.93 %
Proposed thermistor type	GB42JM32H49

Table B.6.3.6.1.2-1: Thermistor Interface Requirements

B.6.3.6.2 Connectors

B.6.3.6.2.1 PLM Side

Table B.6.3.6.2.1-1 identifies the connector type used for the Thermistor interface on the PLM side.

Connector ID nominal	P09
Function	Thermistor Interface
Type nominal	KJL-6T-15-35 PN

Table B.6.3.6.2.1-1: Connector Type PLM side

B.6.3.6.2.2 Instrument Side

Table B.6.3.6.2.2-1 identifies the connector type used for the Thermistor interface on the instrument side.

Connector ID nominal	J09
Function	Thermistor Interface
Type nominal	KJL-3T-15-35 SN

Table B.6.3.6.2.2-1: Connector Type Instrument side

B.6.3.6.3 Pin Allocation

Table B.6.3.6.2.3-1 shows the pin allocation of the Thermistor interface (identical for the PLM and instrument side).

PIN No.	Function	Comment
37	INSTR. THERMISTOR A (S)	
36	INSTR. THERMISTOR A (R)	
35	INSTR. THERMISTOR B (S)	
34	INSTR. THERMISTOR B (R)	
33	ICU THERMISTOR A (S)	
32	ICU THERMISTOR A (R)	
31	ICU THERMISTOR B (S)	
30	ICU THERMISTOR B (R)	
Spare Pins	see ISAICD	
Ground	Shell is Ground	

Table B.6.3.6.2.3-1: Pin Allocation Thermistor interface

Abbreviations:

(S) = Signal (R) = Return

B.6.3.7 Release/Deployment Interfaces

Where an instrument has an I/F to the PLM for release/deployment, the following is defined in the ISAICD.

B.6.3.7.1 Electrical Interface Description

B.6.3.7.1.1 PLM Side

B.6.3.7.1.2 Instrument Side

B.6.3.7.2 Connectors

B.6.3.7.2.1 PLM Side

B.6.3.7.2.2 Instrument Side

B.6.3.7.3 Pin Allocation

B.6.4 Measurement Data Interface

The measurement data interface transfers measurement data from the various instruments to the Formatting and Multiplexing Unit (FMU).
The data transfer on LBR interfaces to the FMU shall be performed via data links that provide a channelized concept supporting the following operational combinations

- instrument nominal or redundant with FMU nominal
- instrument nominal or redundant with FMU redundant.

Cross-strapping shall be performed within the instrument.

B.6.4.1 Electrical Interface Description

The electrical measurement data interfaces are described in figures B.6.4.1.-1 to 5 and tables B.6.4.1.-1&2.

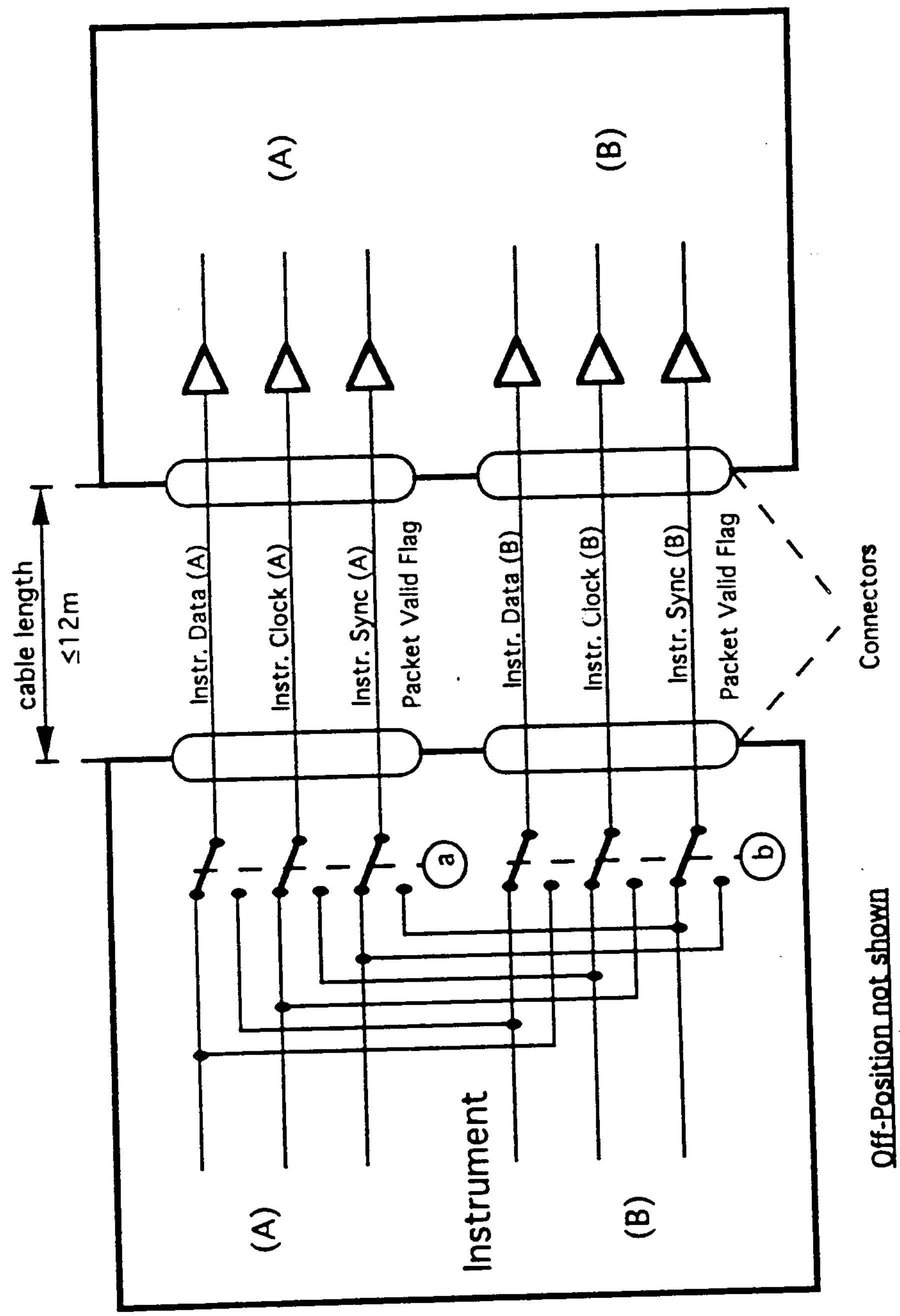


Figure B.6.4.1-1: Measurement Data Interface Crosscoupling

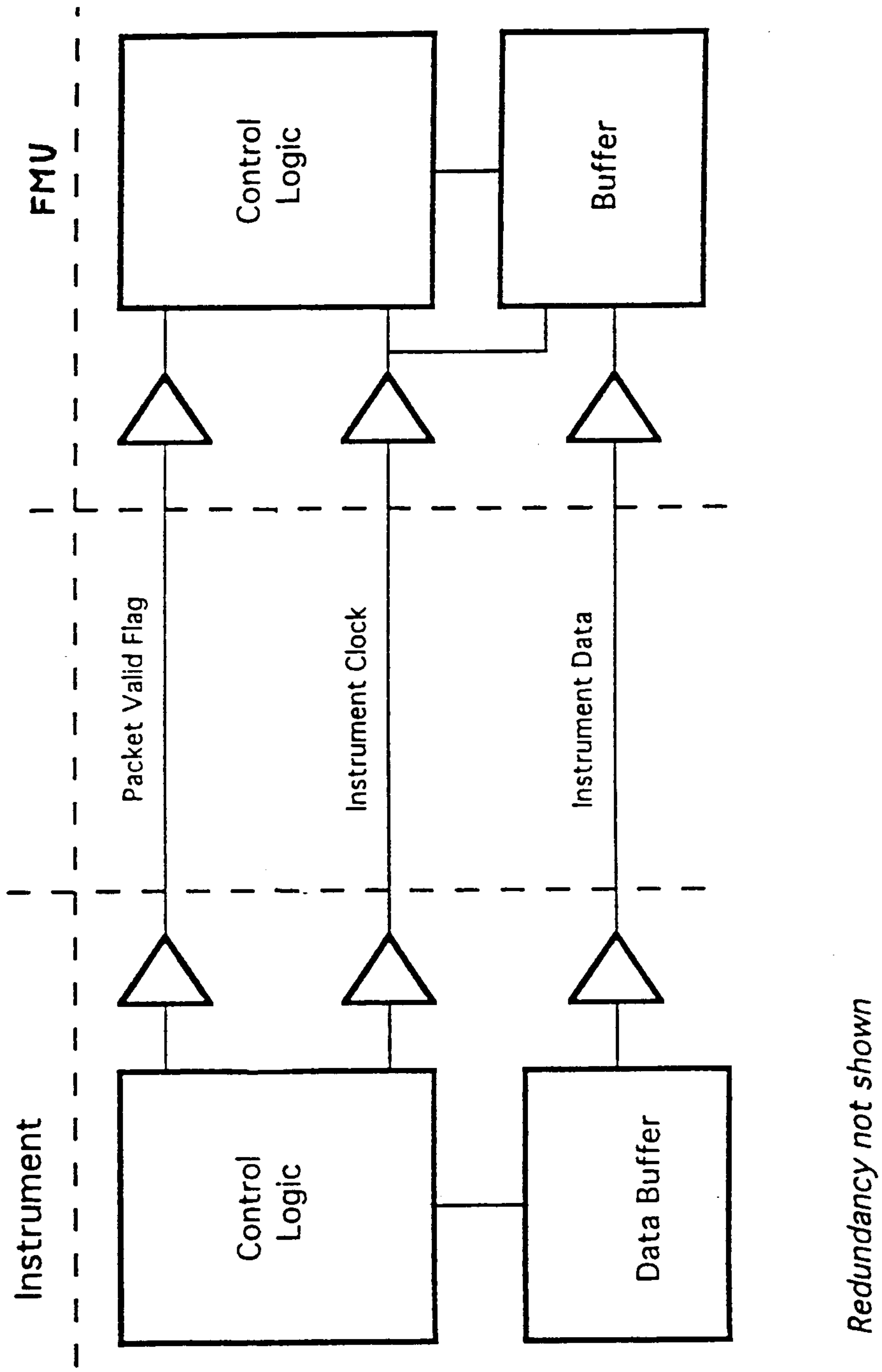


Figure B.6.4.1-2: Measurement Data Interface Blockdiagram

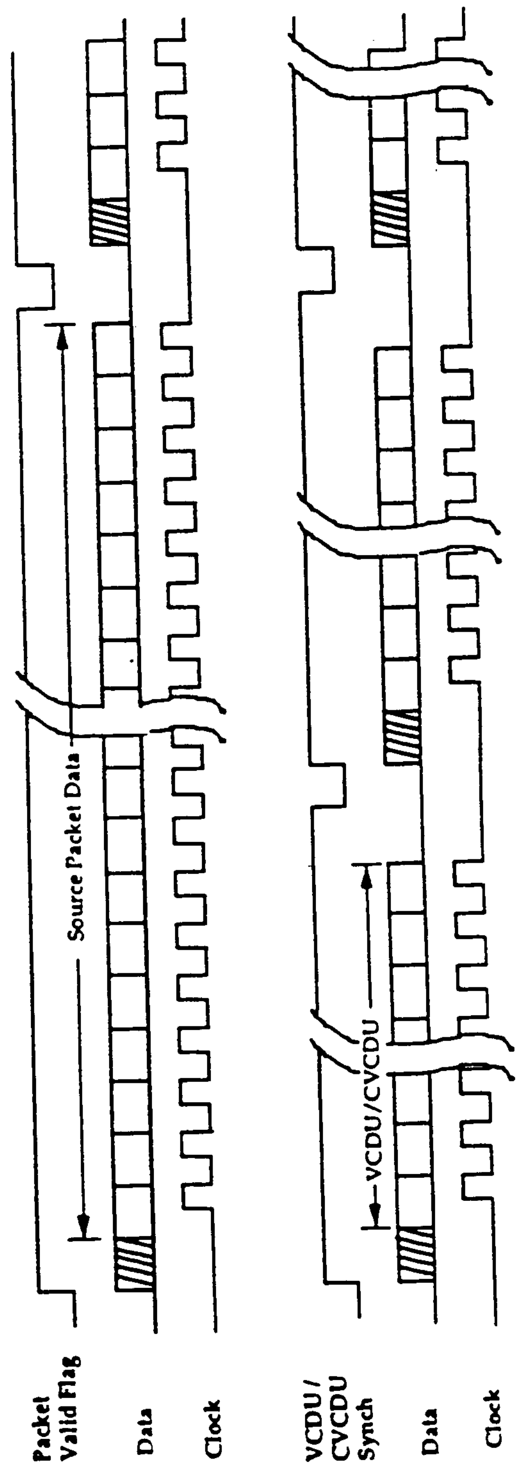
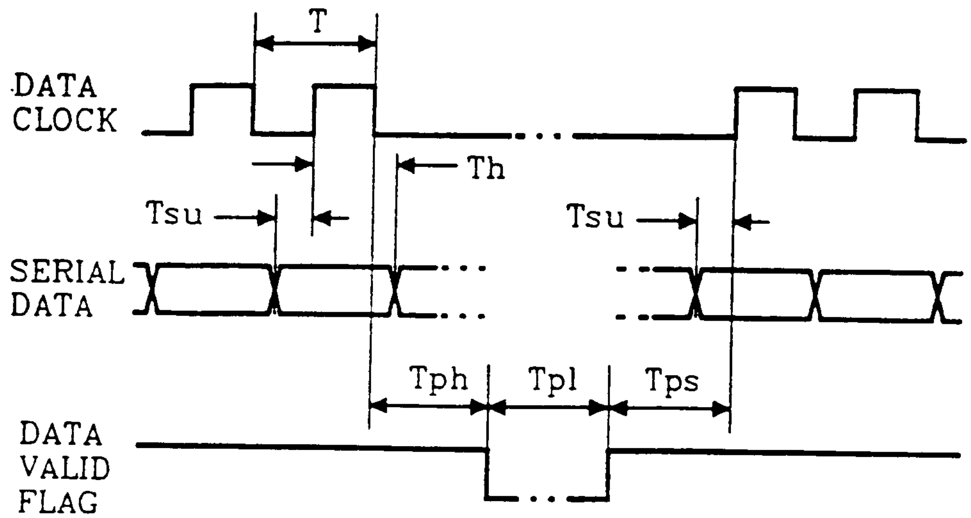


Figure B.6.4.1-3: LBR Timing Schematics



TIME	DESCRIPTION	MAX	MIN
T	DATA CLOCK PERIOD		100 ns
Tsu	DATA VALID BEFORE CLOCK EDGE		40 ns
Th	DATA VALID AFTER CLOCK EDGE		10 ns
Tph	VALID FLAG HIGH AFTER LAST CLOCK	T + 30 ns	T
Tpl	DURATION OF VALID FLAG LOW		T
Tps	VALID FLAG HIGH BEFORE FIRST CLOCK	T + 30 ns	T
Tr	RISE/FALL TIMES - DATA AND CLOCK	15 ns	
Tji	PHASE JITTER - DATA TO CLOCK	± 3 ns	

Figure B.6.4.1-4: LBR Timing

INTERFACE DATA SHEET		(page 1 of 2)
I/F Designation	Low Bit Rate Instrument	Code: LBR

LBR Interface consist of either:

- Packet Data Transmission Interface
  - o Data — LBR Instrument -> HSM
  - o Clock — LBR Instrument -> HSM
  - o Packet Valid Flag — LBR Instrument -> HSM

or:

- VCDU/CVCDU Data Transmission
  - o Data — LBR Instrument -> HSM
  - o Clock — LBR Instrument -> HSM
  - o VCDU/CVCDU Synch — LBR Instrument -> HSM

SOURCE CIRCUIT SPECIFICATION

Circuit Type	Differential AM 26LS 31	
Transfer	DC coupled	
Zero Reference	Signal ground	
Data Encoding	NRZ-L	
Data Rate Range	0 to 10 Mbps	
Active Level Va	$2.45\ V \le Va \le 5.0\ V$	
Quiescent Level Vq	$0\ V \le Vq \le 0.55\ V$	
Logic state "1"	True output high ; complement output low	
Logic state "0"	True output low ; complement output high	
Diff. Voltage ($V_{true} - V_{compl}$)	$1.5\ V \le Vd \le 5\ V$	
Drive capability (true & compl.)	$I \ge 1$ mA at $V_o \ge 2.45$ Volt $I \ge 20$ mA at $V_o \ge 1.3$ Volt	
Sink capability (true & compl.)	$I \ge 1$ mA at $V_o \le 0.55$ Volt $I \ge 20$ mA at $V_o \le 1.70$ Volt	
	Powered	**Unpowered**
Overvoltage (each output to ground) Capability	$-0.5\ V \le V_o \le 5.5\ V$	$-0.5\ V \le V_o \le 5.5\ V$
Source Impedance	$\ge 1k\Omega$	$\ge 1k\Omega$
Emission	$0\ V \le V_o \le 5.5\ V$	
Impedance	$\ge 50\ \Omega$	
Diff. Over-/Under-shoot	$\pm$ 0.8 Volt	
Short Circuit	To ground (each output) for undefinite time without and line to line, without damage or degradation to the device after removal of short	
Leakage Current (non powered)	$\le 100\ \mu A$ at $0\ V \le V_i \le 5\ V$	
Rise and Fall Times	$\tau \le$ TBS ns (10%-90%) (assuming a load of $3k\Omega \| 30pF$)	
Max. Skew between true and compl.	$\tau \le 25\%$ of Rise/Fall Time	

NOTES: Timing shall be verified with 12m twinax cable. Powering up/down shall not cause spikes on the output (RESET level to be maintained), when loaded with a powered receiver

Table B.6.4.1-1: LBR Interface Characteristics

INTERFACE DATA SHEET		(page 2 of 2)
I/F Designation	Low Bit Rate Instrument	Code: LBR
RECEIVER CIRCUIT SPECIFICATION		
Circuit Type	Differential AM 26 LS 32	
Dif. Input Voltage ($V_{true}-V_{compl.}$)	$V_d \geq 1$ Volt	
Req. Input Current (true or compl.)	≤ 1 mA at $V_d \leq 5$ Volt	
Input Capacitance Line to Line	≤ 20 pF	
Overvoltage (each input to ground)	Powered	Unpowered
Capability Source Impedance Emission Impedance	$-0.5\ V \leq V_o \leq 5.5\ V$ $\geq 50\ \Omega$ $0\ V \leq V_o \leq 5.5\ V$ $\geq 1\ k\Omega$	$-0.5\ V \leq V_o \leq 5.5\ V$ $\geq 50\ \Omega$
Short Circuit	Line to Line; Line to GND without damage or degradation to the device after removal of short	
Leakage Current (non powered)	$\leq 850\ \mu A$ at $0.5\ V \leq V_i \leq 5.5\ V$ without deviation from RESET condition	
Diff. Over-/Under-shoot	± 0.8 Volt	
Rise and Fall Times	$\tau \leq$ TBS ns (10%-90%)	
Max. Skew between true and compl.	$\tau \leq 40\%$ of Rise/Fall Time	
HARNESS SPECIFICATION		
Wiring Type Core to Core Cap. Core to Shield Cap.	twinax GORE 6509, REV. 4 30 pF/m ± 10% 55 pF/m ± 10%	
NOTES: -		

Table B.6.4.1-2: LBR Interface Characteristics (cont'd)

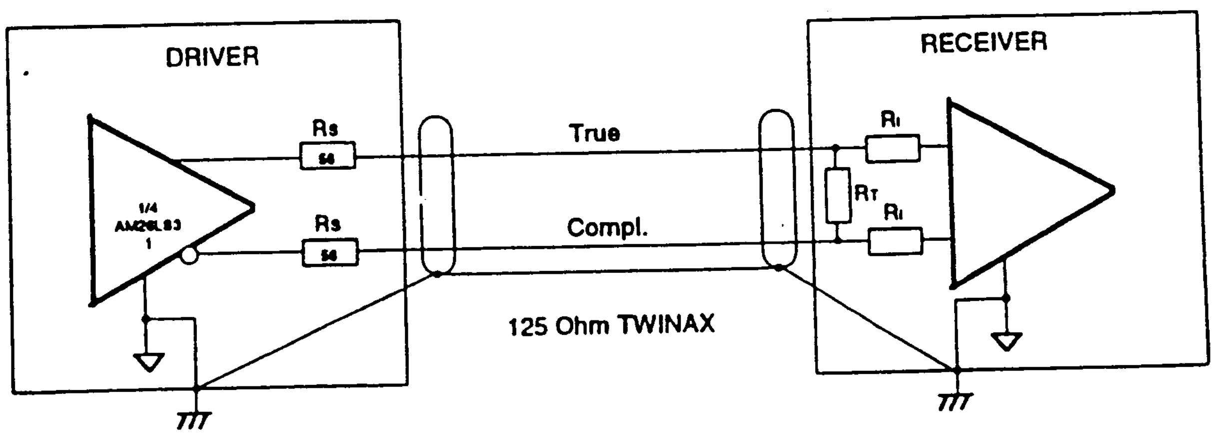


Figure B.6.4.1-5: LBR Interface Arrangements

B.6.4.2 Connectors

TBD

B.6.4.3 Pin Allocation

TBD

B.7 EMC

B.7.1 EMC Design

See Part A.

B.7.2 EMC Performance

B.7.2.1 Spacecraft Power Quality

Each instrument connected to the primary power bus of the spacecraft must be compatible with the power quality as described below:

B.7.2.1.1 Primary Power Source Impedance

The primary power bus impedance (source impedance) as seen from the particular instrument connector bracket is shown in fig. B.7.2.1-1. For all conducted emission measurements on primary power an adequate network simulating the source impedance shall be used.

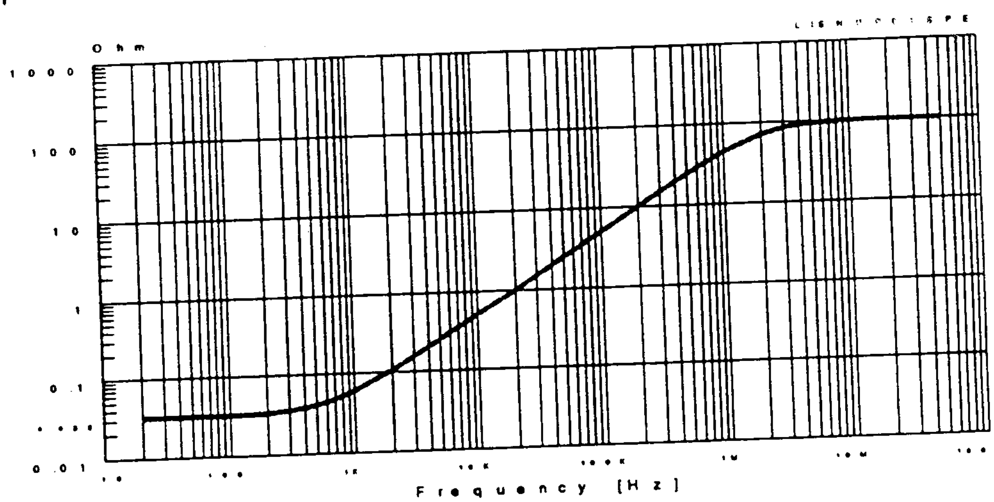


Fig. B.7.2.1-1 Source Impedance of the 28V Unregulated Primary Power Bus

B.7.2.1.2 Primary Power Voltage Characteristics

nominal voltage:	28 V
nominal voltage range:	22 V - 37 V
voltage ripple	1.5 Vpp, measured in BW > 50 MHz
transients	± 1.5 Vpp with 20 ns - 1 µs pulse width

Switch-Off Voltage
At instrument switch-off the voltage transients superimposed on the primary power bus shall not exceed the limits given in fig. B.7.2.1.2-1.

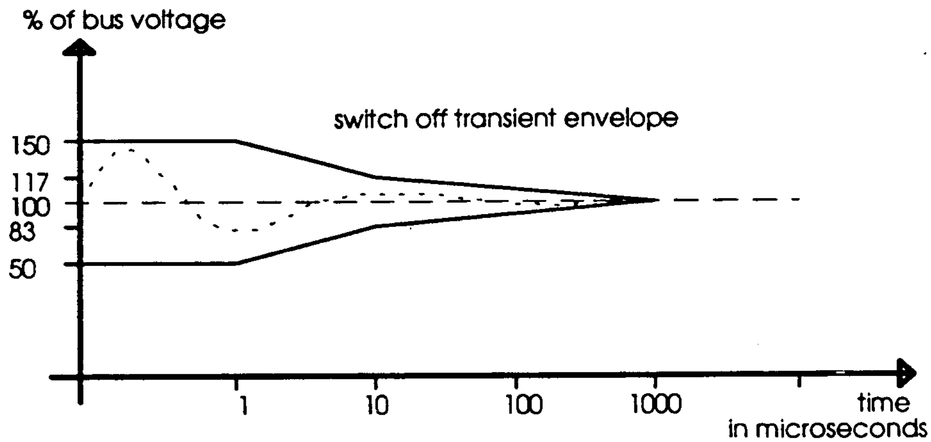


Fig. B.7.2.1.2-1 Switch-Off Voltage Transient Envelope

Load Current Steps

- At a load current step increase (equipment switch-on) of 30 A (TBC) occuring in < 50 µs with di/dt≤ 2 A/µs the primary power voltage is allowed to fall by a level of ≤ 5V for 200 µs max. duration.
- At a load current step reduction of 30 A (TBC) occuring in < 50 µs with di/dt ≤ 2 A/µs the primary power voltage is allowed to rise by a level of ≤ 5V for 200 µs max. duration.

Sun-Eclipse
During sun-eclipse transitions, the rate of any voltage level change will not exceed 1 V/s.

Fault Conditions
Under fault conditions (e.g. bus overload, such as fuse blowing events) the bus voltage will not exceed 42 Volt or fall below 0 Volt. Recovery to nominal operating range will be within a maximum duration of 50 ms.

B.7.2.2 Conducted Emissions

<u>a) Ripple</u>
On the primary power bus the conducted voltage emissions between PLUS and RET leads in time domain shall be ≤ 500 mVpp in a measuring bandwidth of ≥ 50 MHz.
The conducted current emissions in time domain on the primary power bus PLUS and RET leads shall be ≤ 200 mApp in a measuring bandwidth of ≥ 50 MHz.

<u>b) Mode Switching Transients</u>

- Conducted voltage transients measured between PLUS and RET leads of the primary power bus during mode switching (ON/OFF switching excluded) shall be ≤ 1 Vp.
- Conducted current transients measured on PLUS and RET leads of the primary power bus during mode switching (ON/OFF switching excluded) shall be ≤ 1 Ap.

<u>c) Emissions in Frequency Domain</u>
Conducted current emissions in Narrowband (NB) in the frequency range 20 Hz - 50 MHz appearing on the instruments primary power lines between PLUS and RET (differential mode) shall be in the limits of fig. B.7.2.2-1 . .

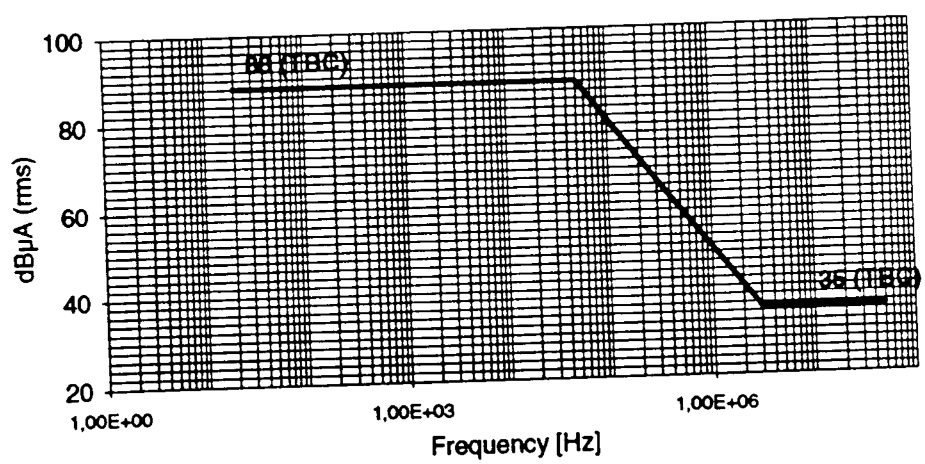


Fig. B.7.2.2-1 CE on Primary Power Lines, NB, DM

B.7.2.3 Conducted Susceptibility

Each instrument shall perform within specification accepting 1 Vrms to be injected on the primary unregulated 28 Volts power line in the frequency range of 20 Hz to 50 MHz.

B.7.2.4 Radiated Emissions

See part A

B.7.2.5 Radiated Susceptibility

See Part A.

B.7.2.6 Electrostatic Discharge (ESD)

See Part A.

B.8 RFC Design

See Part A.

B.9 EGSE

B.9.1 EGSE Overview

B.9.2 EGSE Interfaces

B.9.3 EGSE to Instrument Interfaces

B.9.4 EGSE Infrastructure / Facilities Requirements

B.9.5 EGSE Handling / Transportation / Storage

B.10 Ground Operations